NASA Technical Memorandum 4170

The Office of Exploration FY 1989 Annual Report

Exploration Studies Technical Report Volume II: Space Transportation Systems

(NASA-TM-4170-Vol-2) EXPLORATION STUDIES TECHNICAL REPORT. VOLUME 2: SPACE TRANSPORTATION SYSTEMS Annual Report, FY 1989 (NASA) 210 p

N93-71137

Unclas

Z9/91 0138004

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National Aeronautics and Space Administration
Lyndon B. Johnson Space Center

REPORT DOCUMENTATION PAGE

1. Report No. TM-4170	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle OEXP EXPLORATION STUDIES TECHNICAL REPORT Vol.		5. Report Date August 1989
		6. Performing Organization Code
7. Author(s)		8. Performing Organization Report No.
9. Performing Organization Name and Address Lyndon B. Johnson Space Center Houston, Texas 77058		10. Work Unit No.
		11. Contract or Grant No.
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, DC 20546		13. Type of Report and Period Covered TM - FY 1989
		14. Sponsoring Agency Code

15. Supplementary Notes

16. Abstract

The Office of Exploration (OEXP) at NASA Headquarters has been tasked with defining and recommending alternatives for an early 1990's national decision on a focused program of human exploration of the solar system. The Mission Analysis and System Engineering (MASE) group, which is managed by the Exploration Studies Office at the Lyndon B. Johnson Space Center, is responsible for coordinating the technical studies necessary for accomplishing such a task. This technical report, produced by the MASE, describes the process that has been developed in a "case study" approach. The three case studies that were developed in FY 1989 include:
1. Lunar Evolution Case Study, 2. Mars Evolution Case Study, 3. Mars Expedition Case Study. The final outcome of this effort is a set of programmatic and technical conclusions and recommendations for the following year's work.

17. Key Words (Suggested by Author(s))

On-orbit processing & Assembly
Trajectories
Automation & Robotics
Nuclear power & Propulsion
Advanced Life Support
Martian Moons
Telecommunications, Navigation and Information Management
Transfer Vehicle
Lunar Science
Mars Transfer
Earth-to-Orbit Transportation
Lunar Mining & Oxygen Production

18. Distribution Statement

Unclassified - Unlimited

Category 91

19. Security Classification (of this report) Unclassified	20. Security Classification (of this page) Unclassified	21. No. of pages	22. Price

Disclaimer Statement

The Exploration Studies Process, as explained in detail in Section 2 of Volume I, was a requirements driven, iterative, and dynamic process developed for case study analysis. This process consisted of three parts: (1) requirements generation, (2) implementation development, and (3) integrated case study synthesis.

During the final step of the process, an integrated mission was developed for each of the case studies by synthesizing the implementations developed earlier into a coherent and consistent reference mission. These are presented in Section 3 of Volume I of this annual report. Given the iterative and dynamic nature of this process, there are two important items to note:

- **The integrated case studies do not always reflect a mission that has a direct one-to-one correspondence to the requirements specified in the March 3, 1989, *Study Requirements Document.* Many changes were made to these requirements prior to and during the synthesis activities when warranted.**

- **The integrated case studies presented in Volume I represent the results of the synthesis process. Volumes II, III, and IV are the implementation databases from which the integrated case studies were derived. Therefore, the implementations outlined in Volumes II, III, and IV are generally reflected in the integrated case studies, but, in some cases, the implementations were changed in order to be effectively included in the integrated case studies. These modifications are only briefly discussed in Volumes II, III, and IV.**

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1.0 INTRODUCTION

The three Case Studies set by the Office of Exploration for the Fiscal Year 1989 (FY89) include two Mars scenarios and one Lunar scenario. In response to the Study Requirements Document (SRD), the Transportation Integration Agent (TIA) has developed point designs for vehicles to provide the transportation necessary to accomplish these scenarios. Because each case study was defined in considerable detail, including trajectory types, payloads, and envelope ΔV's, the designs generated are necessarily applicable only to each scenario specified. No effort was allocated for creating new scenarios, although certain design alternatives were examined that would require changes in the baseline scenario. For example, the effect on initial mass in low Earth orbit (IMLEO) was calculated for alternative mission launch years and trajectory classes.

The scenarios are summarized at the top level in Table 1-1. All scenarios for the FY89 Case Studies employ split strategies, i.e., major cargo is carried on unmanned launches that occur at distinct opportunities from the manned ("piloted") missions. No "all-up" missions are baselined. More detailed information on each case may be found in the individual case study detailed descriptions (Sections 2.0-4.0) and in Appendix E.

After completion of the main study, the Mission Analysis and System Engineering (MASE) team synthesized new versions resulting in vehicle changes. These later mission and transportation concepts are reflected in Section 5.0, and are not considered in their respective case studies, Sections 2.0 through 4.0. The changes for Lunar Evolution included major increases in carrying capacity of the Lunar Transfer Vehicles (LTV) and major use of the expendable mode for early missions. In the Mars Expedition, an all-up approach was selected. For Mars Evolution, landers were made expendable

Table 1-1 Summary of Case Studies for Fiscal Year 1989

	Lunar Evolution (CS 4.1)	Mars Evolution (CS 5.0)	Mars Exped (CS 2.1)
Crew size	8	5, 7	3
Staging node	SSF	dedicated new LEO node	none
Reusability of piloted vehicles	yes	yes	no
Aerobrake			
Use	at Earth	Mars, Earth	Mars only
Type	low L/D	low L/D	L/D = 1.0
Orbit at target body	LLO	Phobos (Gateway)	LMO
Trajectories			
Cargo	from SSF	Conjunction	Conjunction
Piloted	from SSF	Opposition & Conjunction	Sprint
ETO pipeline constraints			
Wet	570 t/yr	570 t/yr	570/yr
Dry	90 t/yr	90 t/yr	—
Propulsion			
Cargo	Chemical NEP, 2014	Chemical NEP, 2014	Chemical —
Piloted	Chemical	Chemical NTR, 2014	Chemical
In situ Resources Utilization	LLOX	Phobos LOX	none

and launch opportunities, vehicle sequencing, and cargos were changed significantly. The MASE synthesis versions are presented in greater detail in Volume I

Within each of the case study sections, the same format is followed to provide the greatest ease in locating appropriate material of interest. This report is necessarily limited to top level and only certain detailed information. Additional information is provided in the appendices.

2.0 LUNAR EVOLUTION CASE STUDY (CS 4.1)

2.1 CASE STUDY OVERVIEW

2.1.1 Program Objectives

The major intent of the Lunar Evolution Case Study was to examine methods of providing routine transportation to the moon for both cargo and humans. Plans for the early phases of lunar transportation up through the fully operational phase with a lunar surface base have been prepared. For changes made to the following case study, during the MASE synthesis effort, see Section 5.2.1.

2.1.2 Missions

The main requirement for mission planning is that the program be initiated in 2004 and that two crews of four astronauts each is the maximum rate of personnel transfer into LEO per year.

2.1.3 Requirements

The top level requirements for the case study consist of support for a lunar base during its three phases of buildup—Outpost (human-tended), Experimental (up to 8-crew), and Operational (up to 30-crew base). All the vehicles are intended to be reusable, with the transfer vehicles leaving from the Space Station node and returning to LEO from the lunar vicinity via aeroassist.

The vehicles defined for the case study are intended to provide transportation for humans and cargo from Space Station to lunar orbit and then to the lunar surface with landing vehicles. Eventual use of lunar produced oxygen for propellant is to be a factor in the vehicle development.

Figure 2.1.3-1 illustrates the vehicles that have been defined as well as their domains of operation. Table 2.1.3-1 lists each vehicle and the pertinent requirements from the SRD that drive the vehicle design. Requirements are analyzed futher in Appendix E.

Table 2.1.3-1 Lunar Evolution Vehicles

LPV (Lunar Piloted Vehicle) Crew 8 (emergency, 9), 2 t cargo capacity to moon. Chemical propulsion. Direct Earth entry capability (emergency) 5 g/cm· radiation shield.	LEO <—> LLO
LCSV (Lunar Crew Sortie Vehicle) Crew 8. 2 t cargo (down only). H/O chemical propulsion. No radiation shield.	LLO <—> LSurf
LCV (Lunar Cargo Vehicle) Same propulsion as LPV. 20 t + 3.6 LH, from LEO—>LLO	LEO <—> LLO
LCL (Lunar Cargo Lander) 20 t cargo (down only), H/O chemical propulsion.	LLO <—> LSurf
LPT (Lunar Propellant Tanker) Transports LLOX to LLO	LSurf <—> LLO

2.1.4 Assumptions

Several initial starting points in vehicle design were used in order to reduce the number of design trades that might apply. Many of the vehicle design assumptions are consistent with previous work in OTV Phase A Studies and other space transportation studies. These include the usage of low lift-to-drag and foldable flex-fabric aerobrakes for aerocapture at earth. In addition, on-orbit removal and replacement was assumed for the main engines and main propellant tanks. The vehicles were sized such that their maximum diameter would not exceed a 10 meter launch shroud envelope. Additional details on the assumptions used in this study are given in Table 2.1.4-1.

The aerobrake mass factor is the ratio of the mass of the Earth-return aerobrake to the mass of the largest vehicle being aerocaptured (before adding the braking system).

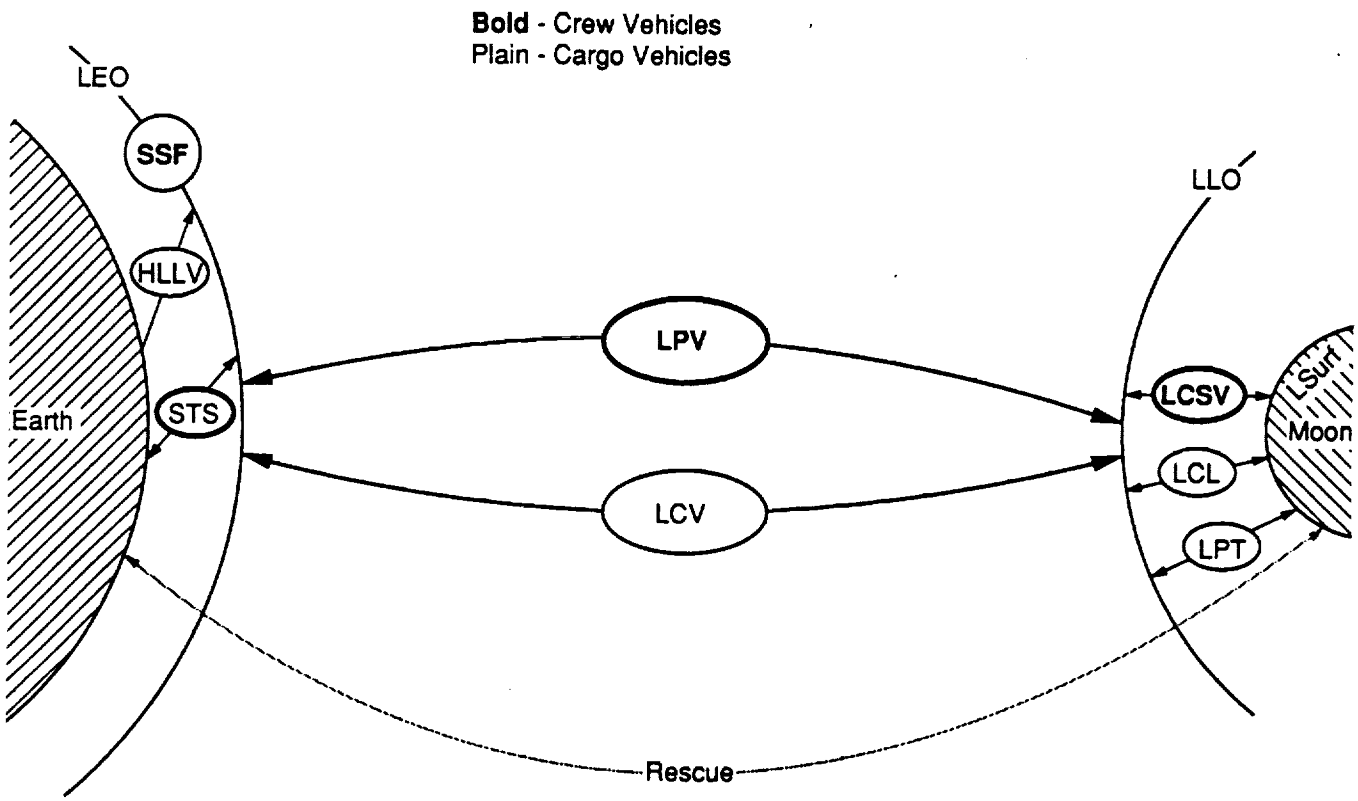


Figure 2.1.3-1 Lunar Evolution—Case Study 4.1—Transportation Vehicles

Table 2.1.4-1 Assumptions for Lunar Evolution Case Study

- Size all crew vehicles for crew of 8, with emergency capability for 9
- Near-term Advanced Cryo-engines (RS-44 class) for LCV and LPV:
  I_{sp} = 481 lb_f-s/lb_m, 15 klb, thrust per engine
- Near-term Advanced Cryo-engines for LCSV, LCL, and LPT:
  I_{sp} = 465 lb_f-s/lb_m, 15 klb, per engine, modified to 15:1 throttleability
- Aerobrakes are
  - low L/D (0.14)
  - foldable, flex-fabric, with point design (10.9% aerobrake mass factor)
- Boiloff: 3.73%/mo. for combined LH_2 and LOX
- Meteoroid and orbital debris shielding included on all vehicles for short missions only
  (Freedom hangars provide protection between missions; blanket or hangar over vehicle for long-term storage on moon)
- Propellant: 2% reserves for I_{sp}; 1.5% residuals; 5% ullage
- Single periapsis burn for LCV and LPV for both TLI and TEI events
- LCV and LPV operations at LLO are with delivered payloads attached
- LPV habitat mass = 10 t for 8 crew nominal, 9 crew emergency
- LCSV cab = 4 t for crew of 9
- On-orbit engine and tank replacement if needed during servicing/refurbishment
- Tankage: 30-mil Al-Li 2090 alloy, plus 15-mil Al-Li bumper spaced 5 cm
  (with 2.5 cm MLI in gap)
- Strive for 10 m diameter P/L envelope (to minimize HLLV requirements)

2.2 VEHICLE DESCRIPTIONS

2.2.1 Configurations

The transfer vehicle intended to deliver humans to and from lunar orbit from LEO is the Lunar Piloted Vehicle (LPV) and is shown in Figure 2.2.1-1. The LPV mission is to deliver a crew of eight to lunar orbit in addition to 2 metric tons of payload. The LPV then returns the crew back to LEO from lunar orbit. Figure 2.2.1-2 illustrates the LPV crew module for the delivery and retrieval of crew. This module is sized to hold a maximum of 9 crewmembers—8 plus a contingency for one more. The LPV crew module is capable of performing a re-entry at Earth independent of the LPV if for some reason the LPV is unable to complete a nominal mission. The LPV crew module also is intended to provide radiation protection for the crew during their trip to and from the lunar vicinity.

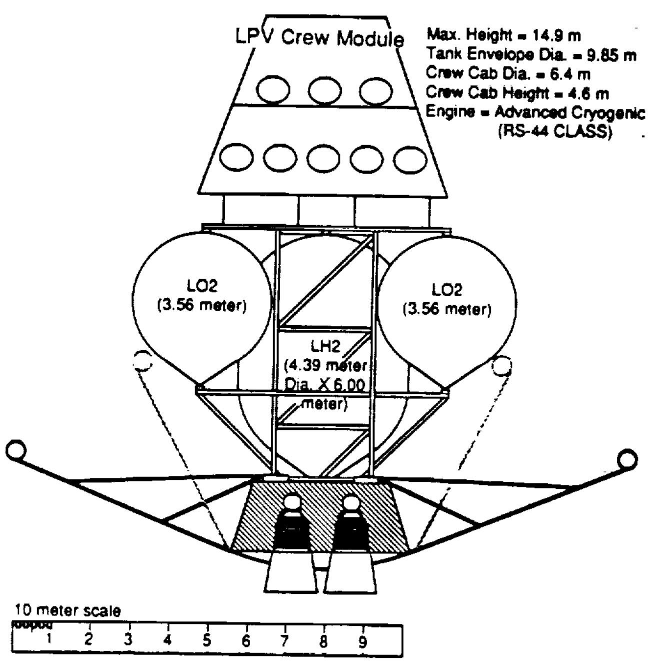


Figure 2.2.1-1 Lunar Piloted Vehicle

The Lunar Crew Sortie Vehicle (LCSV), shown in Figure 2.2.1-3, is intended to deliver a crew of 8 plus 2 metric tons to the lunar surface and then return the crew back to lunar orbit. This lander has four main engines that are required to throttle down in order to accommodate the landing. Figure 2.2.1-4 shows the LCSV crew module overall dimensions.

The Lunar Cargo Vehicle (LCV), Figure 2.2.1-5 is intended to deliver 20 metric tons to lunar orbit from LEO and then return itself to LEO. The LCV uses the identical propulsion system, aerobrake, etc. (the same transfer stage design) as the LPV. This is a favorable commonality situation due to the similar propellant requirements, domain of operation, thrust level requirements, etc.

A Lunar Cargo Lander (LCL), Figure 2.2.1-6, is the vehicle intended to deliver the 20 metric ton payload (brought to lunar orbit from LEO) to the lunar surface. Similar to the LCSV, the LCL uses four advanced cryogenic space engines to perform the landing and ascent at the moon. Depending on whether the LCL begins its mission in lunar orbit (reused or expended), or fully loaded with LOX from the surface, the propellant load required is approximately 25 metric tons. A variation from the LCL that may be used, depending upon the transportation requirements for oxygen from the moon, is the Lunar Propellant Tanker (LPT), Figure 2.2.1-7. This vehicle is configured by simply adding a set of tanks to the top of the LCL for delivery of propellant from the lunar surface to lunar orbit or, alternatively, L1.

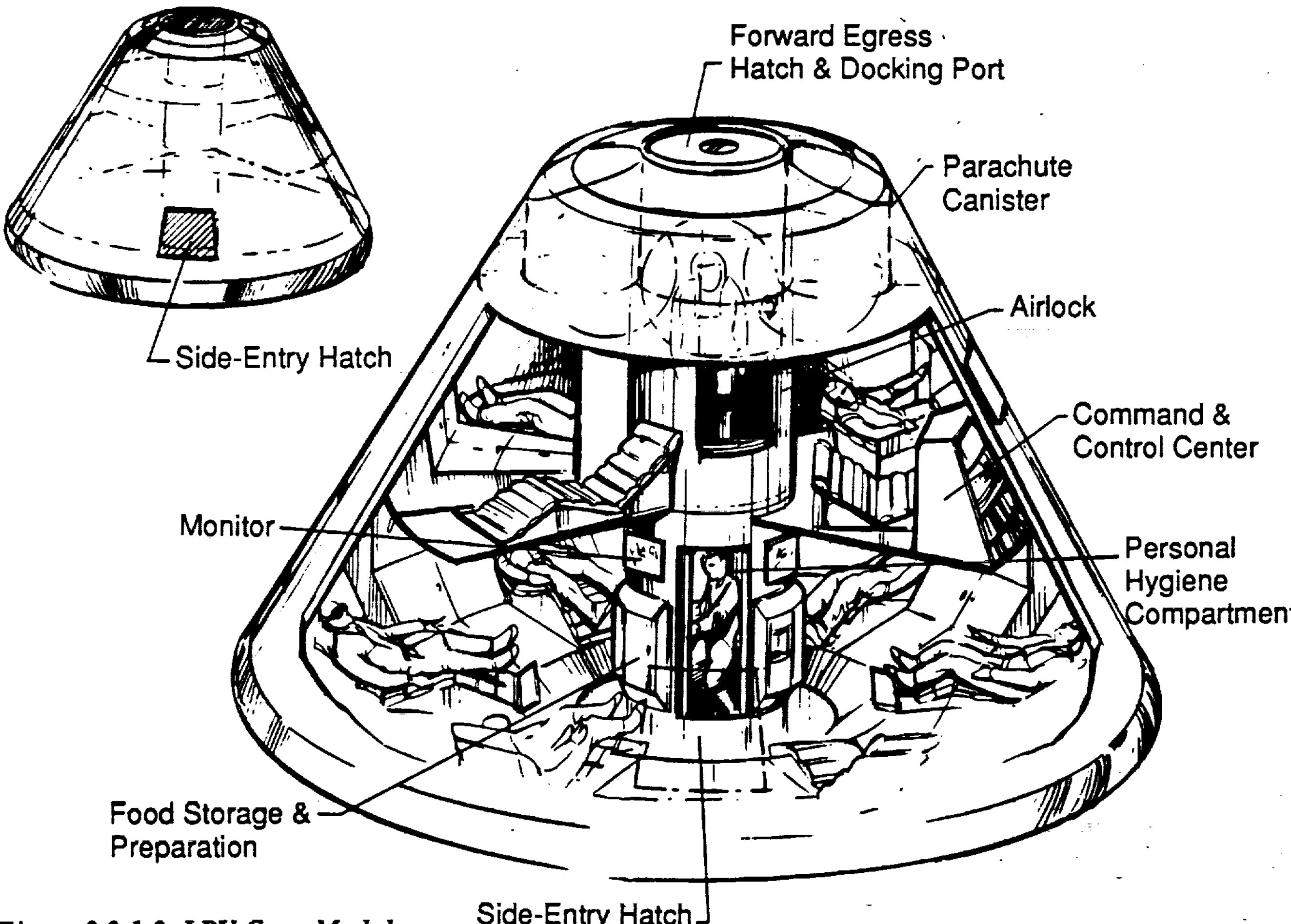


Figure 2.2.1-2 LPV Crew Module

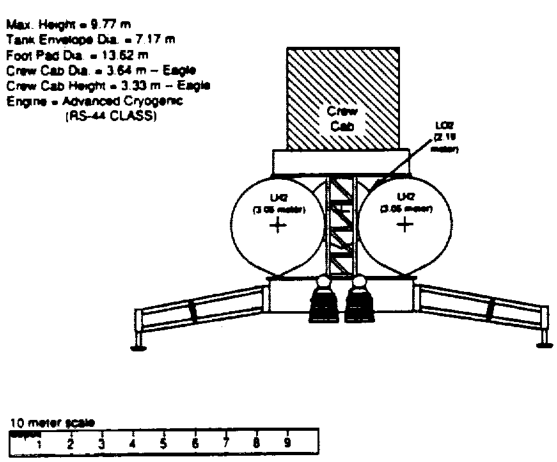


Figure 2.2.1-3 Lunar Crew Sortie Vehicle

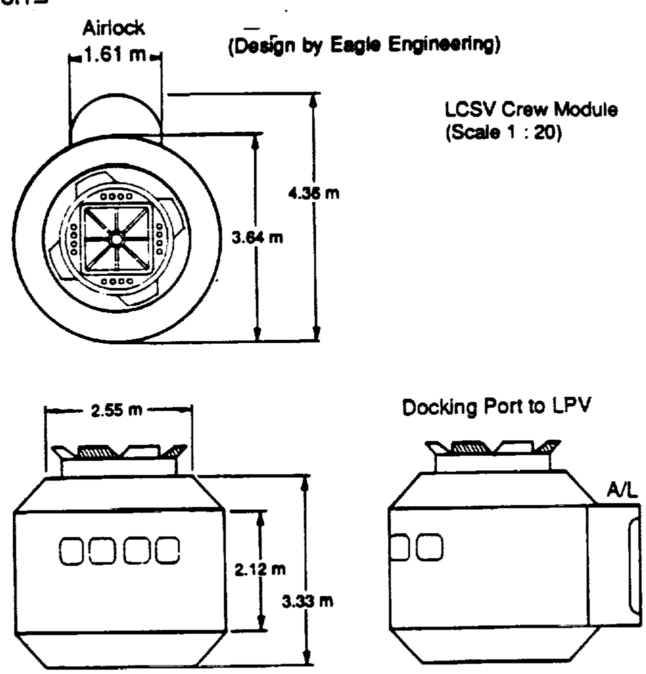


Figure 2.2.1-4 LCSV Crew Module, Plan View

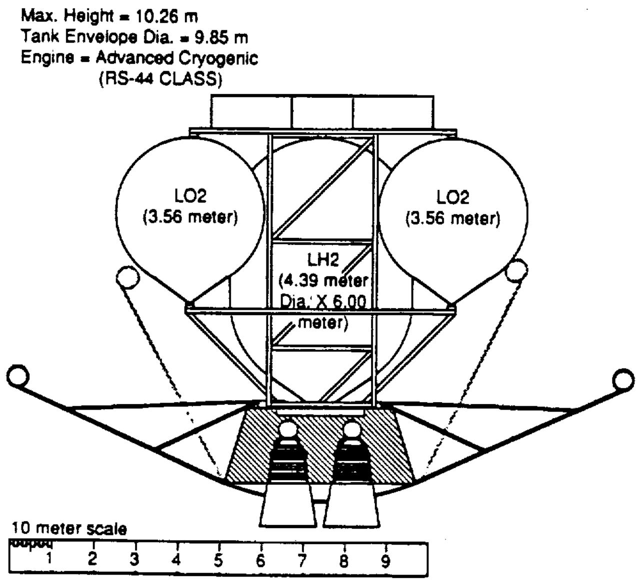


Figure 2.2.1-5 Lunar Cargo Vehicle

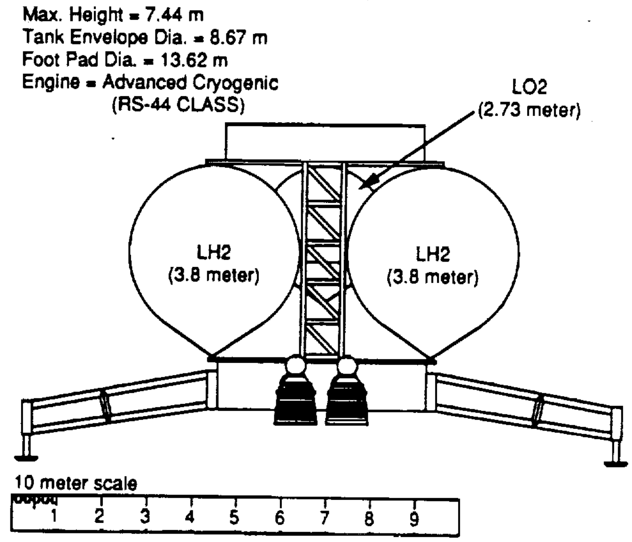


Figure 2.2.1-6 Lunar Cargo Lander

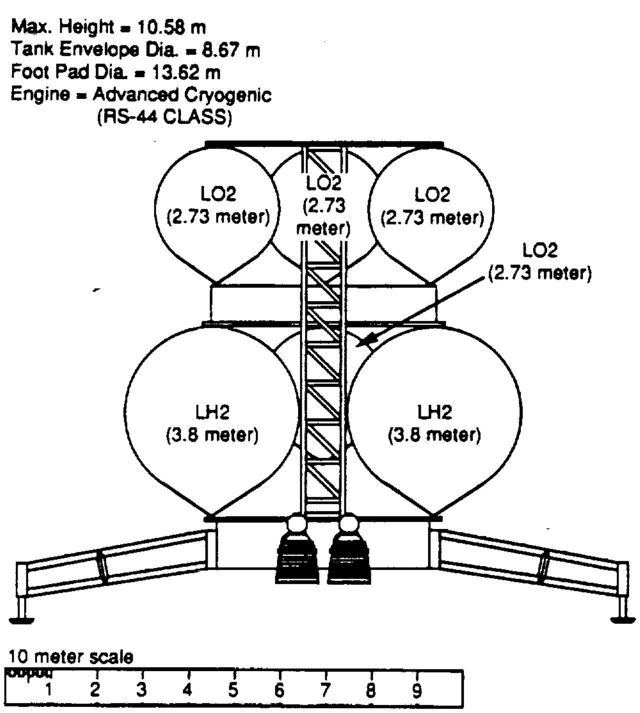


Figure 2.2.1-7 Lunar Propellant Tanker

Table 2.2.1-1 provides a summary of the lunar vehicles with their dry masses and their loaded propellant capacity. Also addressed are the areas for potential commonality. Detailed mass allocations are provided in Appendix B.

Table 2.2.1-1 Lunar Evolution Vehicles

Vehicle	Dry Mass (kg)	Propellant Capacity (kg)	Commonality
LCV	5530	59090	Entire Propulsion System — LPV
LPV	5530	59090	Entire Propulsion System — LCV
LCSV	2884	11950	Same Engines, Legs, Avionics as LCL
LCL*	3360	25000	Entire Propulsion System — LPT
LPT	4900	25000	Entire Propulsion System — LCL

Conclusions: 3 Total Propulsion System Designs Required

*LCL can deliver 13545 KG to lunar surface in addition to 4 t (crew) round trip to LLO - LSurf - LLO

2.2.2 Element Summaries

The key elements that make up the Lunar Evolution stable of vehicles are shown in Figures 2.2.2-1 through -5.

Dry Mass	5,530 kg
Payload Mass	12,000 kg
Propulsion System	
Propellant Type	Chemical-LH_2/LOX
Engines	
Number	2
Type	RS-44 Class
Mass (ea.)	210 kg
Thrust (total)	133.4 kN (30 klbf)
I_{sp} (481 sec)	4.71 kN-s/kg
Propellant Mass	59,090 kg
Initial T/W	0.249
Mass Fraction	0.90
Total Mass (includes wet payload)	76,620 kg

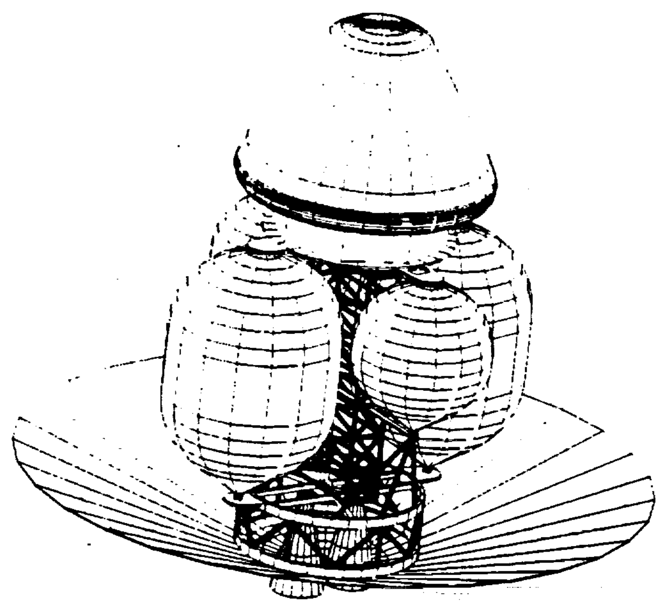


Figure 2.2.2-1 Lunar Piloted Vehicle

Dry Mass	2,884 kg
Payload Mass	6,000 kg
Propulsion System	
Propellant Type	Chemical-LH_2/LOX
Engines	
Number	4
Type	RS-44 Class
Mass (ea.)	210 kg
Thrust (total)	266.9 kN (60 klbf)
I_{sp} (465 sec)	4.55 kN-s/kg
Propellant Mass	11,950 kg
Mass Fraction	0.82
Total Mass (includes wet payload)	20,834 kg

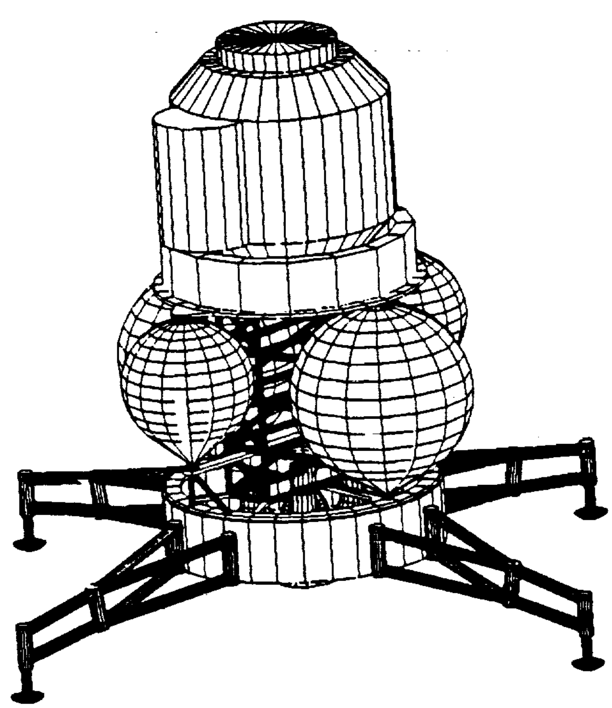


Figure 2.2.2-2 Lunar Crew Sortie Vehicle

2.2.3 Commonality

Achievement of a high degree of commonality among transportation systems was made possible by the fact that the cargo-carrying capabilities are compatible with the masses needed to provide crew

Dry Mass	5,530 kg
Payload Mass	20,000 kg
Payload Volume (cyl. - 8m dia., 10m ht.)	
Propulsion System	
Propellant Type	Chemical-LH_2/LOX
Engines	
Number	2
Type	RS-44 Class
Mass (ea.)	210 kg (461 lbs)
Thrust (total)	133.4 kN (30 klbf)
I_{sp} (481 sec)	4.71 kN-s/kg
Propellant Mass	59,090 kg
Initial T/W_E	0.22
Mass Fraction	0.90
Total Mass (includes wet payload)	84,620 kg

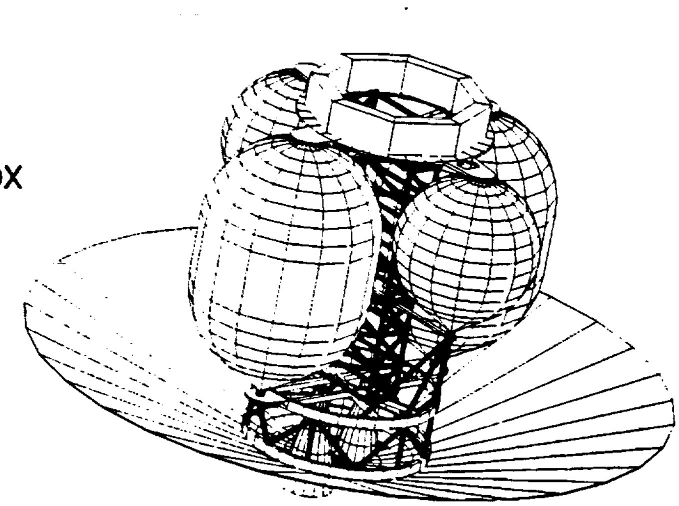


Figure 2.2.2-3 Lunar Cargo Vehicle

cabin facilities for the sizes of crew selected. In addition, it was found practical to employ common landing systems, central truss support structures, and Earth-return aerobrakes for both the cargo and piloted vehicles. An optimum approach for maximizing commonality would be to design the crew compartment as a "bolt-on" to the same interface that will be used for cargo. Once the LPV crew module is sized, the cargo delivery capability of the vehicle can be determined. This then determines the amount of equipment and facilities that can be provided per flight of the LCV and LCL.

The flight engines are selected to be common for all lunar vehicles. Engine design is mostly driven by landing requirements, including wide-range throttling, dust resistance, fault diagnostics, and health maintenance. These attributes will enhance the transfer vehicle capabilities, however.

Dry Mass	3,360 kg
Payload Mass	20,000 kg
Propulsion System	
Propellant Type	Chemical-LH_2/LOx
Engines	
Number	4
Type	RS-44 Class
Mass (ea.)	210 kg
Thrust (total)	266.9 kN (60 klbf)
I_{sp} (465 sec)	4.55 kN-s/kg
Propellant Mass	25,000 kg
Mass Fraction	
Total	0.86
Total Mass (includes wet payload)	48,360 kg

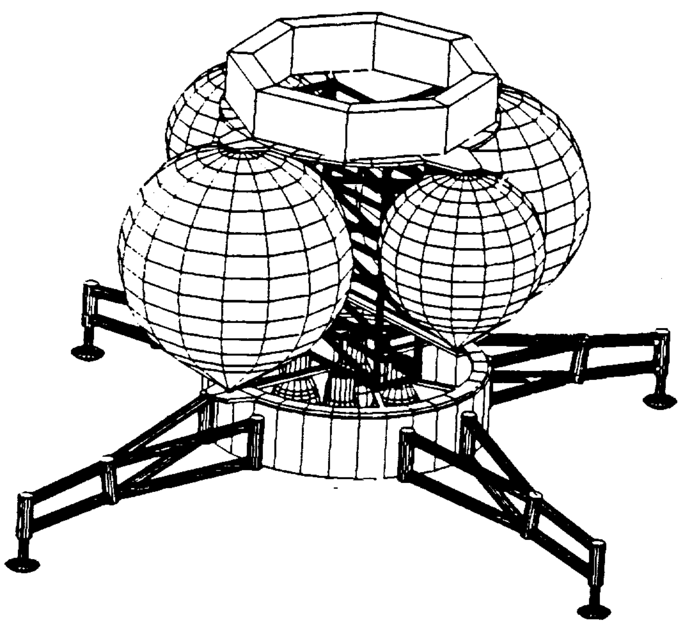


Figure 2.2.2-4 Lunar Cargo Lander

Dry Mass	4,900 kg
Payload Mass	26,280 kg
Propulsion System	
Propellant Type	Chemical-LH_2/LOX
Engines	
Number	4
Type	RS-44 Class
Mass (ea.)	210 kg
Thrust (total)	266.9 kN (60 klbf)
I_{sp} (465 sec)	4.55 kN-s/kg
Propellant Mass	25,000 kg
Mass Fraction	
Total	0.87
Total Mass	
(includes wet payload)	56,180 kg

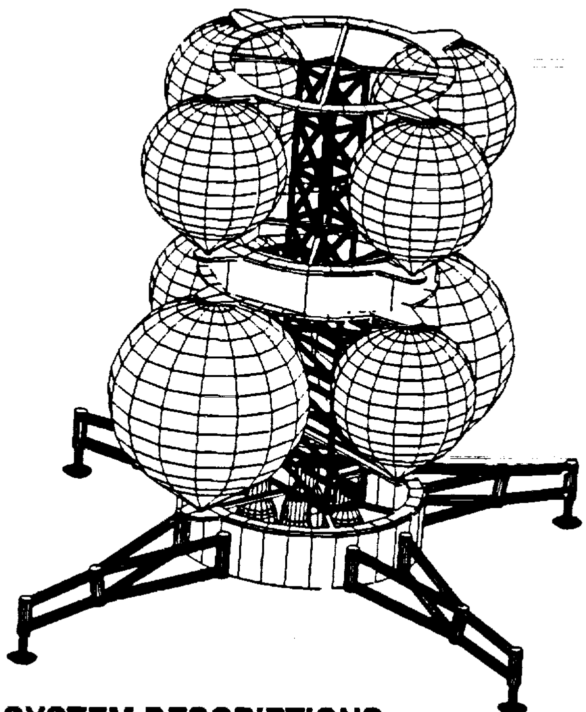


Figure 2.2.2-5 Lunar Propellant Tanker

2.2.4 Cargo Accommodation

The total payload mass for the LCV and LCL cargo vehicles is 20.0 tonnes per flight. Based upon volume and size constraints in the shroud, payloads up to 10-m in diameter can be accommodated.

2.2.5 Science Accommodation

There are no major science requirements for the Lunar Evolution transportation vehicles except to deliver payloads, which may include major amounts of science equipment, to the lunar surface. Because of the short transit times to and from the moon, and the proximity of the moon to Earth, there are only minor needs for spaceborne science. Some lunar surface observations could be flown opportunistically on these manned vehicles, although it is usually more efficient and appropriate that they be on dedicated, long-lived unmanned satellites.

2.3 SYSTEM DESCRIPTIONS

2.3.1 Habitats

The trans-lunar crew module was shown in Figure 2.2.1-2. It is based upon an Apollo-style shape to enable a back-up direct entry capability in the event the aerobrake system is disabled. Crewmembers are arranged radially around a central hollow core which contains a personal hygiene compartment and an airlock for forward egress from the module. This forward lock is also the docking port for shirt-sleeve transfer of crew into the LCSV crew cabin, Figure 2.3.1-1. The latter module also contains a personal hygiene compartment and airlock, but the total volume is much less because occupation of this module is expected to last only from a few hours to perhaps one or two days, whereas the LPV crew module could be occupied for up to 30 days, allowing for the times specified in the SRD for Earth-lunar transits and LLO operations.

(Design by Eagle Engineering)

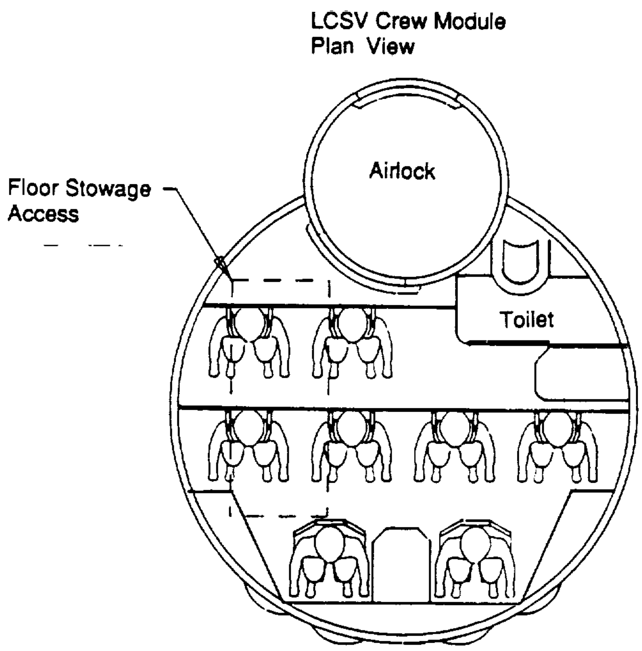


Figure 2.3.1-1 LCSV Crew Module

In both crew systems, the LSS is highly open. Food and water are pre-supplied, although additional drinking water will be available from the on-board fuel cells. The air revitalization system aboard both the LPV and LCSV crew modules consists of oxygen being drawn from the main propulsion system for the on-board requirements and carbon dioxide removal performed by an expendable chemical (LiOH) or regenerable molecular sieve.

2.3.2 Propulsion Systems

The transfer vehicles (LPV and LCV) are both outfitted with two cryogenic space engines. The purpose of this design is to provide one engine-out capability for the trip to and from lunar orbit. The engine selected for the mission is an advanced cryogenic expander cycle engine. A major reason for this choice involves the smaller package compared to existing technology engines due to the higher chamber pressures. Also, long life, high performance, and space serviceability (health monitoring and modularity) are attributes desirable in an engine. The LCSV and LCL both utilize four engines identical to the transfer vehicles but with the additional requirement of wide throttling to accommodate the lunar landing.

2.3.3 Aeroassist Systems

Re-entry velocities for aerocapture into LEO on return from the moon are 11.5 km/s, about the same as Apollo. Use of flexible ceramic cloth for a major portion of the shield reduces aerobrake mass by over 57% from a rigid system (3550 kg for all-rigid compared to 1520 kg for the 15.9 m diameter flex-fabric aerobrake). This in turn leads to a blunt, low lift-to-drag (L/D) aeroshield design. The diameter of the brake is sized to reduce heating to the limits of the selected flexible Thermal Protection System (TPS) cloth. An optimum approach is to use a 4.5 m diameter central core of high temperature ceramic tiles, an outer annulus of flex-fabric Tailored Advanced Blanket Insulation (TABI), and a high temperature resistance graphite polyimide support frame. This system would be deployable on-orbit by command of built-in electromechanical actuators. More information on aerocapture brake designs is provided in Appendix D.

2.3.4 Communication Systems

A major groundrule of the lunar missions was that mission operations would be controlled from the ground (Earth). Communications links will be needed between all elements and Mission Control on Earth in order to provide critical data for operations and decision making for crew and hardware safety.

Critical events such as crew transfers, propellant transfers, and rendezvous/docking will require television coverage. These will be required particularly for the cargo vehicles to permit ground control, but also as a backup for manned vehicles. Two channels will be required per vehicle (stereo, color, high resolution). Because there will be two vehicles involved in most critical operations, this results in four channels at any given time for a total required data rate of 4.15 Mbps. One meter diameter dish antennas with 90 W power are anticipated to fulfill this requirement. Continuous voice and engineering data contact is also a requirement. Video and data communications for base operation, production facilities, and emergencies is required. These Lunar Base links are expected to consume considerably more bandwidth than the transportation vehicles.

2.3.5 Power Systems

At least 2000 W. will be provided to operate crew systems. All power will be generated using fuel cells, drawing from allocated cryogens in the H/O propulsion system. About 350 kg of cryogen (less than 0.6% of the total initial propellant load) will be required during a two week occupation of the LPV. Electrical energy storage can be provided by nickel-hydrogen battery technology.

2.3.6 Thermal Systems

The thermal control system for anticipated nominal operations is a passive heat rejection system using sealed heat pipes and body-mounted radiator plates. Emergency cooling is available from the cryogenic propulsion system. The thermal protection system of the aerobrake is via use of passive materials with high insulative and thermal resistance properties (see aerobrake discussion, Section 2.3.3).

2.4 OPERATIONS CONCEPT

2.4.1 ETO Manifest

Definition of the Earth-to-orbit vehicle is beyond the scope of this report. The lunar vehicle dry masses are all below six tonnes, which is well within the lift capability of the Shuttle or the larger ELVs. A new HLLV will be needed for two reasons, however. First, the lunar vehicles are designed to fit within a 10-m diameter shroud, which is an ETO capability that currently does not exist. Second, to launch the large quantities of cryopropellant will require a greater lift capability if large numbers of launches are to be avoided.

2.4.2 On-orbit Assembly

No on-orbit assembly is required for these vehicles. It will be necessary to perform in-space cryopropellant transfers, however.

2.4.3 Mission Operations and Sequences

Throughout the phases of lunar evolution, the operations of the vehicles may change depending upon the transportation systems capabilities and available infrastructure. Figures 2.4.3-1 and -2 illustrate the various phases of lunar evolution for the transportation system. The first cargo flight to the moon requires a LCL loaded with propellant to be delivered to lunar orbit along with a payload. The LCL is then expended on the surface after delivering the payload. The first crew mission is similar, but the LCSV must return to lunar orbit after going to the surface. Then the LCSV is de-orbited to crash into the moon.

During the interim cargo and crew flights to the moon (before there is propellant or servicing capability available at the moon), the LCL and LCSV vehicles must be delivered to lunar orbit or stationed there from a previous mission. Then, propellant must be delivered to lunar orbit (in addition to the payload/crew) to resupply the landers. This is based upon the assumption that the landers could operate with minor servicing in lunar orbit (man-tended) or be returned to LEO if major equipment malfunctions arise. Servicing of equipment is discussed in Section 2.4.5.

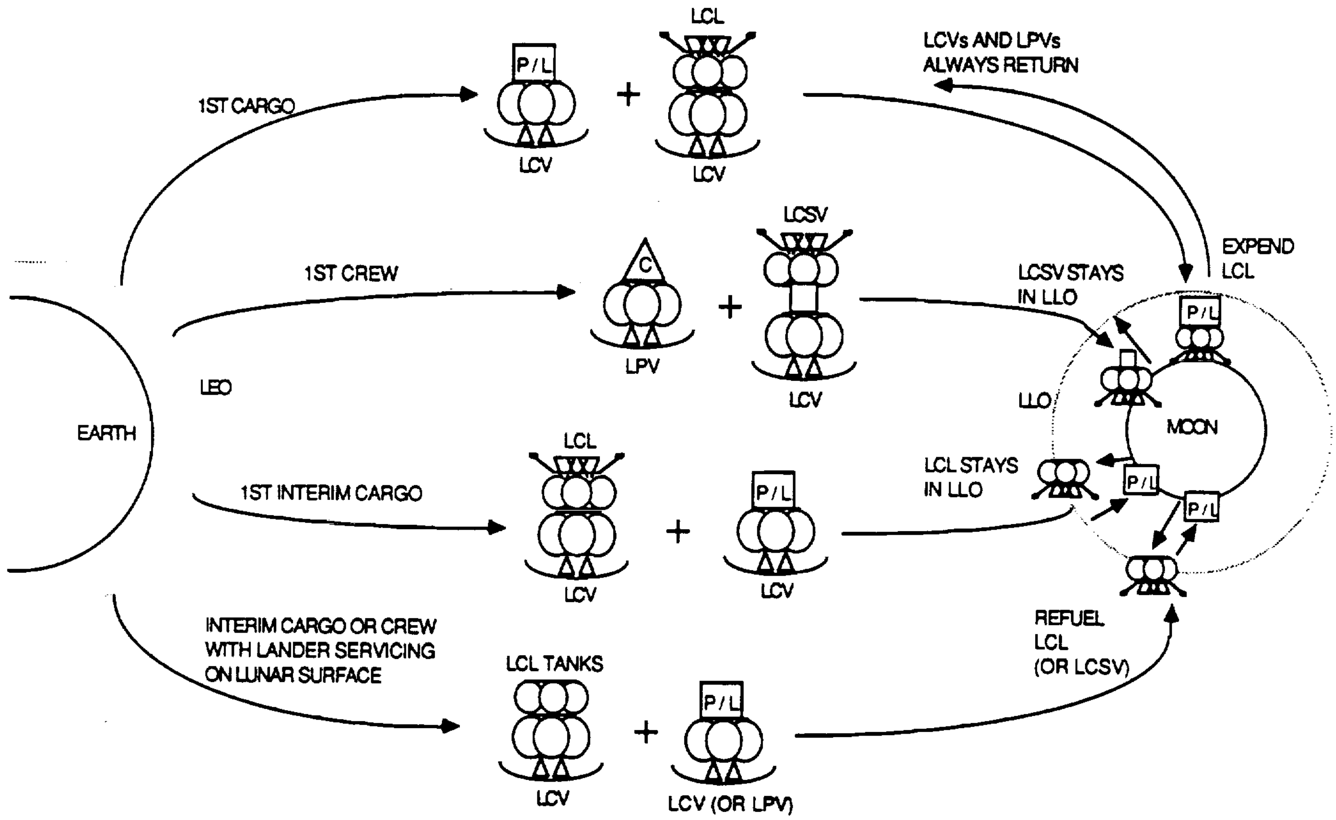


Figure 2.4.3-1 Initial Phases of Lunar Evolution

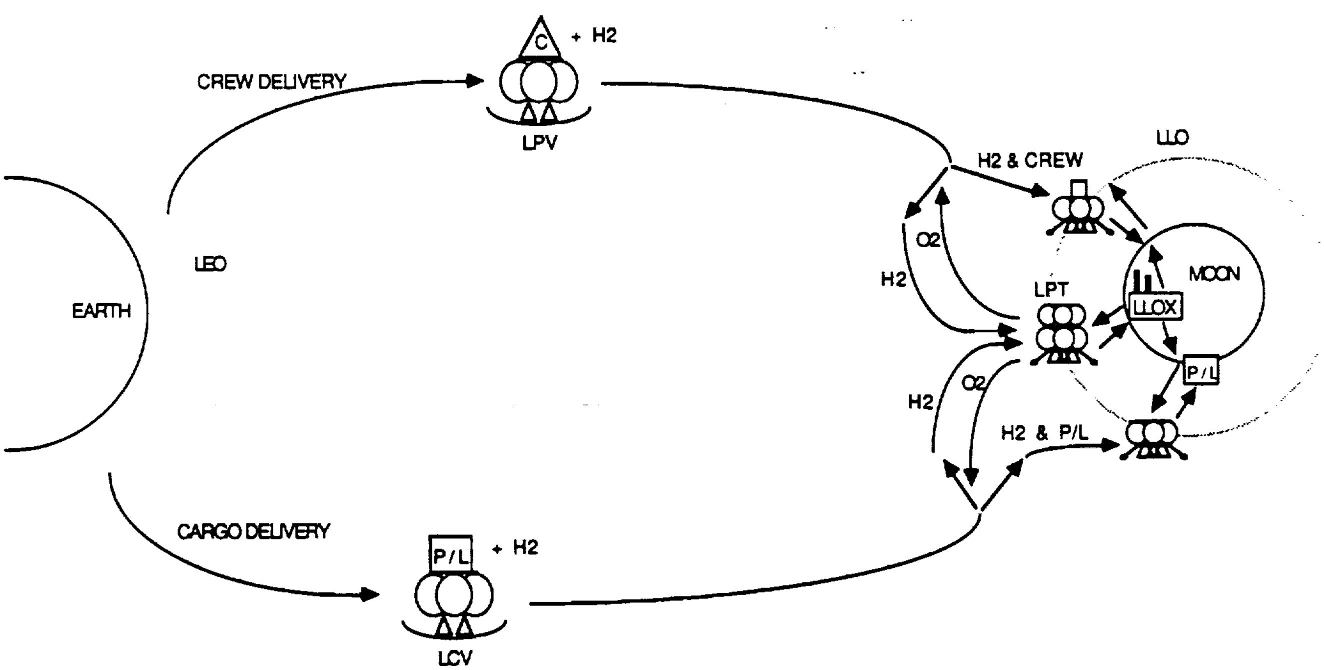


Figure 2.4.3-2 Lunar Propellant Usage with Cryogenic Vehicles

Figure 2.4.3-2 shows the scenario where lunar oxygen and lunar base servicing of the landers exist. The LCL and the LCSV attain their hydrogen from the transfer vehicle as it also delivers either payload or crew to lunar orbit. The transfer vehicle hands off the payload in lunar orbit and then takes on oxygen for the return trip. Landers get their oxygen at the moon surface and are based and serviced there.

Crew Members enter the LCSV from the LPV in lunar orbit via a shirtsleeve access after docking, as shown in Figure 2.4.3-3, for the excursion to the lunar surface.

2.4.4 Reliability and Safety

Reliability is enhanced by providing one engine-out capability for all cryopropulsion systems. This is a measure beyond Apollo, but is appropriate because of the greater risk with the higher performance cryoengines. All other systems are at least dual fault tolerant.

2.4.5 Useful Life

It is estimated that the flex-fabric foldable aerobrake can be rated for a useful life of at least five flights. This is based on a conservative allocation for accumulated damage to the flexible material caused by the turbulance-induced flutter, particle

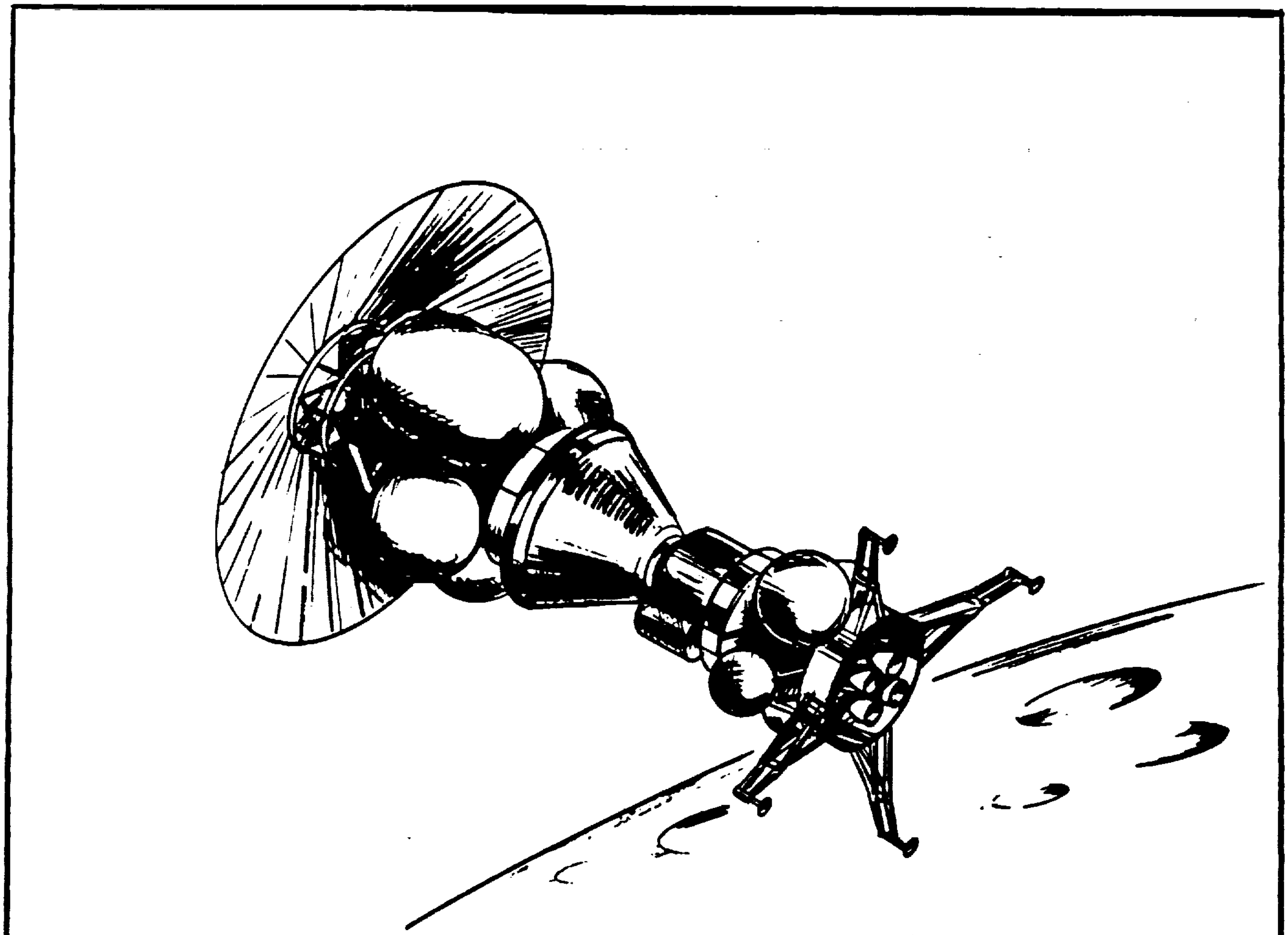


Figure 2.4.3-3 LPV and LCSV Docking for Transfer of Crew

impact, and handling. A rigid brake incorporating brittle tiles may also require periodic replacement or repairs. All other items require less servicing than this, as seen in Table 2.4.5-1. The basis for engine servicing is the anticipated requirement for an advanced engine. A specification for longer re-use could allow an increase in this number. The propellant tank limit is based on a 15-year exposure to micrometeroid flux.

Table 2.4.5-1 Vehicle Servicing Assessment

Vehicle Subsystem/ Component	Expected Life (Missions)	Interfaces for Removal/Replacement: Fluid	Electrical Connectors	Mechanical
Engine	10-20	2 Low Press. Liq. 3 High Press. Gas	1	2 Actuators & Attach Struct.
Propellant Tank	30	1 Low Press. Liq. 1 Low Press. Gas	1	4 Structural Fittings
Avionics	>50	None	Multiple Connectors	Structural Attach
Aerobrake	5	None	None	Structural Attach
Landing Legs	30	None	1	Structural Attach

2.5 DEVELOPMENT SCHEDULES

A development schedule for transportation is shown in Figure 2.5-1. See discussion in section 2.6 on technology needs and section 2.7 on precursors.

2.6 TECHNOLOGY NEEDS

2.6.1 Technical Description

Transfer Vehicle Engine—The need exists for an advanced cryogenic engine in order to perform transfers to and from the moon (or Mars). The appropriate thrust level is between 10 klb_f and 20 klb_f per engine.

Long life in terms of number of starts and total burn time for the engine is an important driver. Engine restart in orbit or on the surface of the moon will be a requirement for an advanced cryogenic engine.

Operation in areas remote from the servicing capabilities on the surface of the Earth will require that reusable engines be space serviceable (replaceable in their entirety, and capable of being purged and cleaned if contaminated), and all engines must have self-contained health monitoring and diagnostics capability. Finally, performance and packaging improvements over existing space engines would provide considerable benefits in reducing the mission propellant requirements by not only increased specific impulse, but by minimizing vehicle dry weights associated with compact configurations.

The major benefits of an advanced cryogenic engine for lunar missions would be in the smaller envelope it would provide for vehicle packaging, and the higher performance the engine could provide. In addition, a space serviceable, health-monitored engine will be essential to a transportation system that involves routine flights to the moon.

Figure 2.6.1-1 shows the differences in the packaging characteristics of an existing engine (product improvement of existing technology) vs the more desirable package resulting from the use of a higher technology engine. The smaller engine envelope is due to the higher chamber pressure of the advanced engine. The net difference in vehicle configuration is significant not only geometrically, allowing a 3.3m length reduction, but the dry mass increase for the longer vehicle must also be considered.

Chemical Descent Engine—The major drivers in selecting engines for descent and/or ascent from either the moon are propulsion system reliability and performance; in that order. The trade-off decision is in weighing the benefits of higher performance against possible corresponding penalties in system reliability. This typically amounts to a comparison between a presumably inherently reliable propulsion system (perhaps pressure-fed storable bipropellant) and the higher performing pump-fed propulsion concepts (storable or cryogenic).

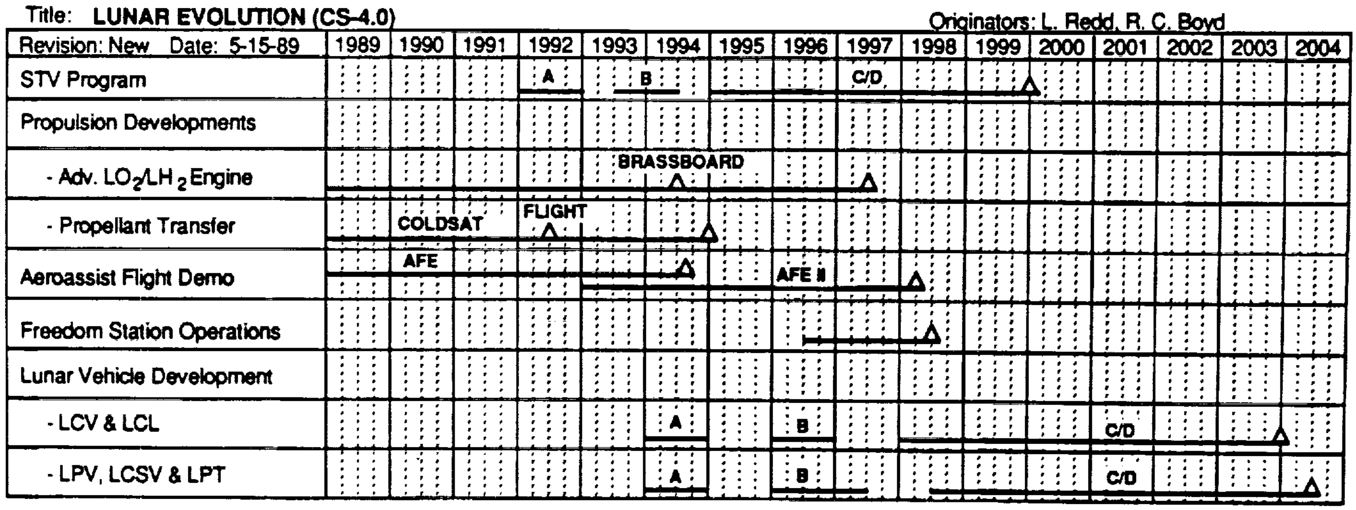


Figure 2.5-1 Development Schedule for Lunar Evolution (CS 4.0)

Advanced engine technology is necessary in order for a cryogenic propulsion system to successfully perform landing and ascent from the lunar surface. For example, in providing the thrust range necessary for descent ignition, hover and final descent, and ascent from the surface, the engine may be required to throttle over a wider range than consistent with present cryogenic engine throttling capabilities because of clustering and engine-out considerations.

In this Case Study, the commonality between the LCSV and LCL has been examined in terms of engine throttling needs, using the Apollo groundrules of 3 m/s^2 (0.31 Earth-g) deceleration for the de-orbit burn and 0.64 m/s^2 (0.065 Earth-g) at touchdown on the lunar surface. In addition, multiple descent engines were assumed to allow for at least one-engine out capability. Adopting the approach that no thrust-vector misalignment is permitted eliminates a two-engine approach and also leads to the need to shutdown an opposing engine if

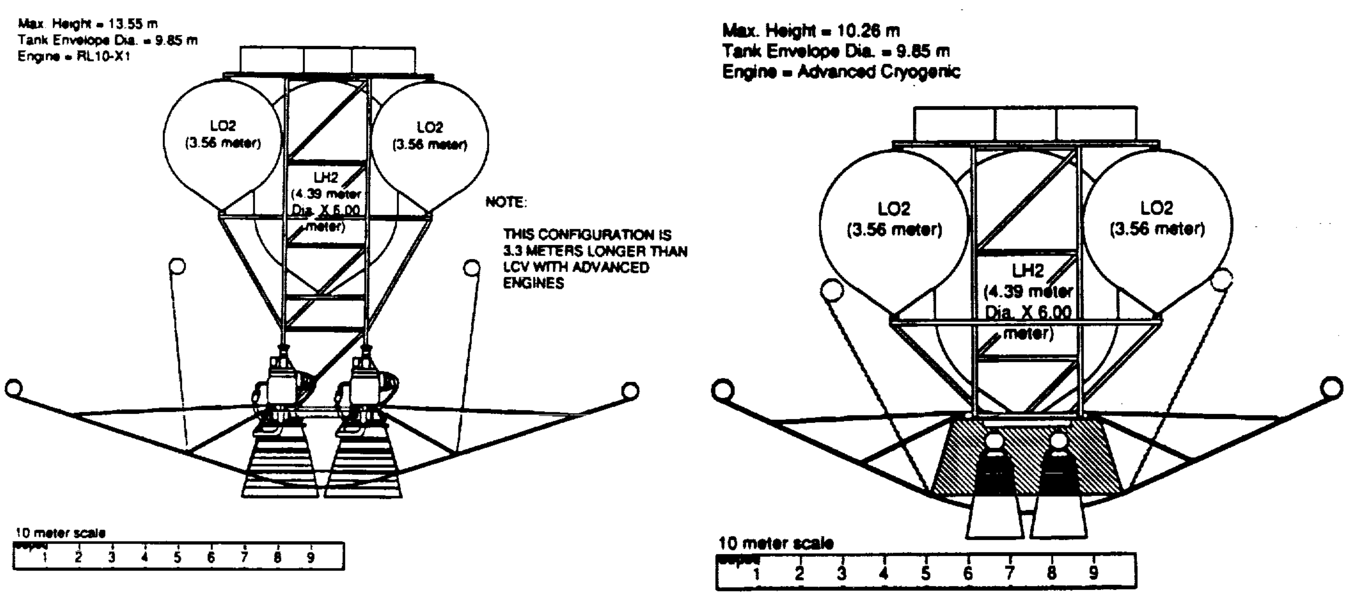


Figure 2.6.1-1 Comparison of LCV Designs with (a) Standard Cryoengines and (b) Advanced Space Engines

an outboard engine is the one to fail. Thus, for both the four and five engine cases, it will be necessary to shut down two of the engines in order to preserve thrusting symmetry. The highest thrust an engine must provide is for de-orbit, wherein an engine is out (and, if appropriate, a second is shut down) and the heaviest vehicle, the LCL, is being decelerated. The minimum thrust an engine must be capable of providing is at lowest throttle, with all engines operating and supporting the lightest landed weight vehicle, the LCSV. Note that the ratio of initial to touchdown mass is just above two for the two vehicles, but for the ratio between them is 5.44. This factor of commonality thus causes an additional factor of about two needed in the throttling range. If the LPT were included in this requirement, the range would be even higher. Values for various engine combinations are given in Table 2.6.1-1. It is seen that the optimum number of engines to minimize the required throttling ratio is four, but that the required range is about 35, which is normally considered beyond that practical for a single engine. One solution is to use smaller engines, but in larger arrays. Another solution would be to reduce the cargo lander payload or to increase the crew sortie vehicle's cargo payload so as to more closely equalize the initial and final masses of the two vehicles (the LCSV is more than a factor of two lighter).

Table 2.6.1-1 Throttling Assessment for Lunar Landing

Vehicle	Number of Engines	Throttling Ratio
LCSV	3	39
	4	35
	5	44
	6	39
LCL	3	15
	4	13
	5	16.5
	6	15

For lunar missions the use of oxygen and hydrogen in the transportation system may provide significant benefits for descent and ascent over the Apollo approach of using storable propellants. In addition to the performance advantages, the ability to take advantage of in-situ produced propellants (oxygen) may provide payoffs to the efficiency of the transportation system.

2.6.2 Need Dates

The advanced space engine is required by mid-1997 as a flight qualified end item for integration into the flight qualification test vehicles.

2.7 PRECURSOR NEEDS

2.7.1 Data

High resolution imaging under varying sun elevation angles is needed for landing site assessments. In addition, scientific data on chemical composition and geologic structure are needed for exploration goals as well as for gauging the potential for resource utilization.

2.7.2 Infrastructure

A heavy lift launch vehicle and an in-space servicing node are needed to support operations of the Lunar Evolution scenario. A space transfer vehicle (STV) development program preceding the lunar development program will help alleviate schedule and technical risk, while providing a broadening of the overall space infrastructure.

2.7.3 Demonstrations

Results from the Aeroassist Flight Experiment (AFE) demonstration is needed by 1994 to support Phase B studies and permit analysis of data before initiation of Phase C/D. A second development and demonstration program, AFE II, is also shown in the schedule, with a flight in early 1998 to correlate with the C/D phase starts for the transportation vehicles. Similarly, an extension of COLDSAT development effort should be accomplished before C/D start to prove-out techniques for on-orbit cryopropellant handling and storage. One-year evaluation periods have been allocated between Phase A and B, and between B and C for the lunar vehicle development programs.

2.8 HUMANS-IN-SPACE RESEARCH NEEDS

There are no new requirements for human operations in space in order to support the transportation vehicles. In-flight experience with telerobotics may provide enhanced capability for certain operations, such as propellant transfers.

2.9 TRADE STUDIES

Trades have been performed on new versus advanced cryoengines (see section 2.6.1), with a decision that favors the new engines. The use of LLOX should be restricted to needs for propellant in the lunar vicinity, rather than for needs which arise at Earth. Increasing the mixture ratio of LLOX-to-fuel for the return to Earth does not result in large reductions in propellant needs (<12% for O/F>10). Two-stage TLI produces 7% or less savings in LEO propellant.

2.10 CASE STUDY SUMMARY AND CONCLUSIONS

Due to the smaller crew accommodations that need to be provided for transfer to and from the moon, the Lunar Evolution vehicles are relatively lightweight compared to Mars mission vehicles. A high degree of commonality between piloted and cargo systems was readily achieved because of the selection of cargo payload mass consistent with the needs for crew systems support. This also allows for a crew rescue mode by keeping the cargo system in reserve at the servicing node during piloted missions. Then, using an in-space spare crew module, an emergency situation could be accommodated by mounting the module on the LCV to convert it to an LPV rescue vehicle. This is the recommended approach to providing crew rescue in LLO.

It was found possible to package all vehicles within a 10-m diameter by 15-m long launch shroud. Although the lunar vehicles could be assembled on-orbit from sub-assemblies launched within a 4.5-m diameter payload constraint (e.g., STS or ELV), it is recommended that a large-diameter shroud HLLV be developed to eliminate costly and possibly hazardous LEO operations for assembling the final vehicle.

It is recommended that a one engine-out criterion be adopted for cryogenic liquid hydrogen/oxygen propulsion engines. By employing engine clusters with spare engine capability, a critically needed gain in system reliability is attained. At the same time, if a two engine-out capability were required, the number of engines in the cluster would have to be increased from 4 to 6, with no relief in required thrust performance. For these reasons, a one fault tolerant approach instead of the SRD dual-fault tolerant approach for cryoengines is recommended, as is used in the Space Shuttle Transportation System. Throttling ratio for landing depends directly on groundrules for landing, especially for touchdown acceleration. Engine clustering, with preservation of one engine-out capability causes the required ratio to increase further if the recommended groundrule that no operating engine be purposively shut down during a descent is retained. It is recommended that new analysis be performed of the permissible range of landing closure rates.

Development of a near-term advanced space engine is recommended for lunar missions. Important attributes of this engine would be good specific impulse performance in a small package, retractable nozzle, very wide throttling range, long-lived reusability, built-in health and diagnostic monitoring, and tolerance to lunar dust.

A separate study, applied to a similar but non-identical case as CS 4.1, has shown the disadvantages of attempting to export lunar LOX for use at LEO by chemical propulsion means alone. For this reason, it is recommended that LLOX be utilized for operations between low lunar orbit and the lunar surface, and for return of vehicles to Earth, but *not* for Earth-to-moon transfer. The Lunar Propellant Tanker (LPT) can be eliminated as a separate vehicle. If some LLOX is required to be transported to LEO, it could be accomplished by the LCV (up to 10 t). Use of high oxidizer/fuel ratio engines to accentuate the leverage of LLOX does not have a major payoff, particularly if LLOX is restricted to use in the lunar vicinity.

A direct-to-surface option was evaluated and shown to provide some benefits in reduction of IMLEO for delivery of constant sized payloads to the lunar surface. However, piloted missions did not show this advantage because of the heavier crew module on the LPV compared to the cab on the LCSV. The LPV module is more massive (10,000 kg versus 4,000 kg) because of two main drivers: the need for more space and facilities to accommodate the longer live-in time, and the requirement for back-up direct entry capability, which means incorporation of an ablator aerobrake onto the module. Thus, for many of the same reasons as Apollo, staging in low lunar orbit has benefits in mass reduction. However, if LLOX were available for a direct return flight, staging in LLO would not be advantageous.

3.0 MARS EVOLUTION CASE STUDY (5.0)

3.1 CASE STUDY OVERVIEW

3.1.1 Program Objective

The objectives of the Mars Evolution case study are to establish a Martian moon gateway and to select technologies that will further enhance solar system exploration, especially a permanent facility on Mars' surface. The gateway and technologies' primary objective is to reduce the dependency of space exploration on Earth resources, particularly propellant. Two technologies receiving special focus are tether systems for momentum transfer between vehicles and the gateway moon and water mining at the gateway for cryogenic propellant manufacturing. An additional objective is for significant scientific research to be carried out during all phases of gateway and Mars surface development. Moderating these ambitious goals is an Earth-to-orbit launch mass limit of 570 tonnes per year. The Lunar Evolution case study is limited to the same amount to provide a valid comparison between case studies. Considerable changes to the following case study were made during the MASE synthesis effort. For these changes, see Section 5.2.2.

3.1.2 Missions (Implementation)

The Mars evolution case study defines the first seven missions of an ongoing program starting with a chemically propelled, aerobraked vehicle taking three crew to Phobos and Deimos and ending with seven crew riding a high energy vehicle powered by a nuclear thermal rocket. Figure 3.1.2-1 shows the master schedule of missions and how the various vehicles are used across time. The missions are broken into three phases: Initial science outpost (2004-2008), human-tended (2009-2014), and operational (after 2014). The first three missions explore the Mars system and set up the gateway moon to support the human-tended phase. The next two missions use the gateway to reduce Earth launch masses. And finally, all following missions use nuclear powered interplanetary vehicles and reuse the chemical powered landers based at the gateway from the previous missions.

3.1.2.1 Mission-1: Phobos and Deimos Exploration—As shown in Figure 3.1.2.1-1, the first mission sends three people to explore Mars' moons. They leave on May 31, 2004 on an opposition trajectory with a Venus swingby on November 17, 2004 followed by an aerobrake into Mars orbit on April 10, 2005. From a 250 by 18000 km phasing orbit, two crew depart the Mars Piloted Vehicle (MPV) in the Phobos/Deimos Excursion Vehicle (Ph/DeEV) and spend two weeks exploring Phobos and two weeks exploring Deimos. Meanwhile the remaining crewperson deploys two communication and a surface mapping satellite, twelve surface navigation beacons and a Mars surface rover/sample return package. The rover is teleoperated by the single crewmember in the MDV and samples are gathered and launched back into orbit for retrieval. On July 19 all three leave Mars and transfer directly back to Earth. At Earth they aerobrake the entire vehicle first into a highly elliptical orbit, then again into low Earth orbit (LEO) to rendezvous with the assembly/refurbishment facility that co-orbits with Space Station Freedom. This double-pass aerocapture reduces the g-loads on the crew and vehicle providing lower health risk and lighter structures. If a problem with the MPV had arisen jeopardizing the crew during aerobraking, they would have transferred to an emergency reentry capsule designated as the Earth Crew Capture Vehicle (ECCV) and let the MPV fly-by Earth and be lost. This first mission has an initial mass in LEO (IMLEO) of 637.8 tonnes. This is made up of three 140 tonnes TMI stages, a 203.8 tonne MPV, and a 15.2 tonne AOTPM and Crew Cab that make up the Phobos Excursion Vehicle. The trans-Earth injection stage is an integral part of the MPV.

3.1.2.2 Mission-2: Human Landing on Mars—The second mission sends five crew to Mars to land on the surface. This mission is depicted in Figures 3.1.2.2-1 and 3.1.2.2-2. They depart Earth August 22, 2005 on a conjunction trajectory and aerobrake into a 250 by 33120 km orbit at Mars on February

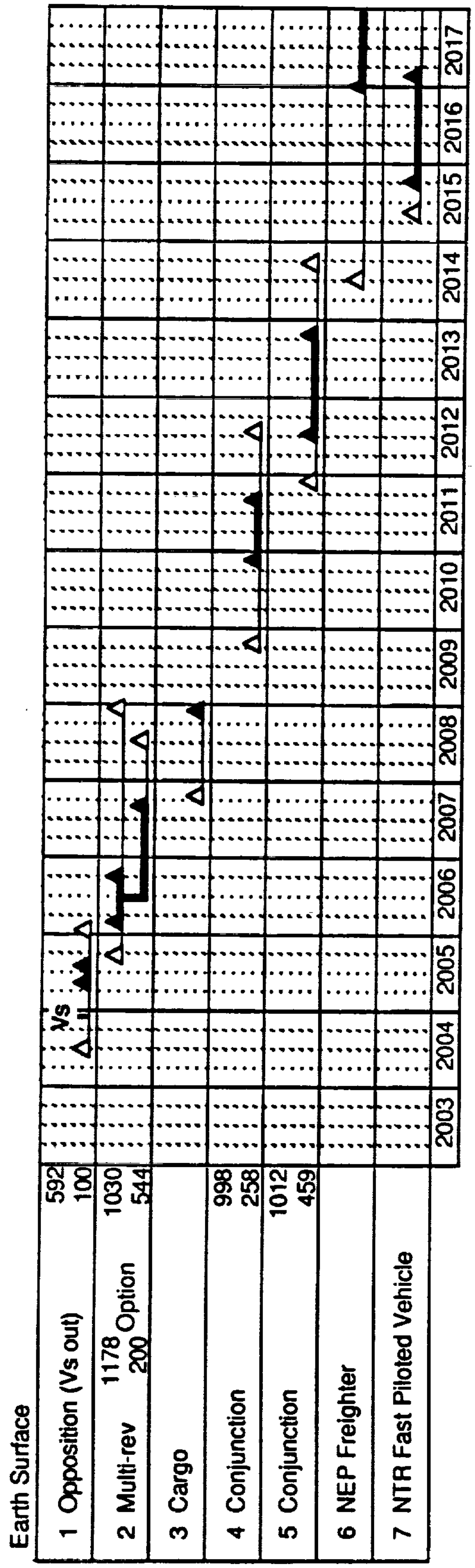


Figure 3.1.2-1 Mars Evolution (CS-5.0)

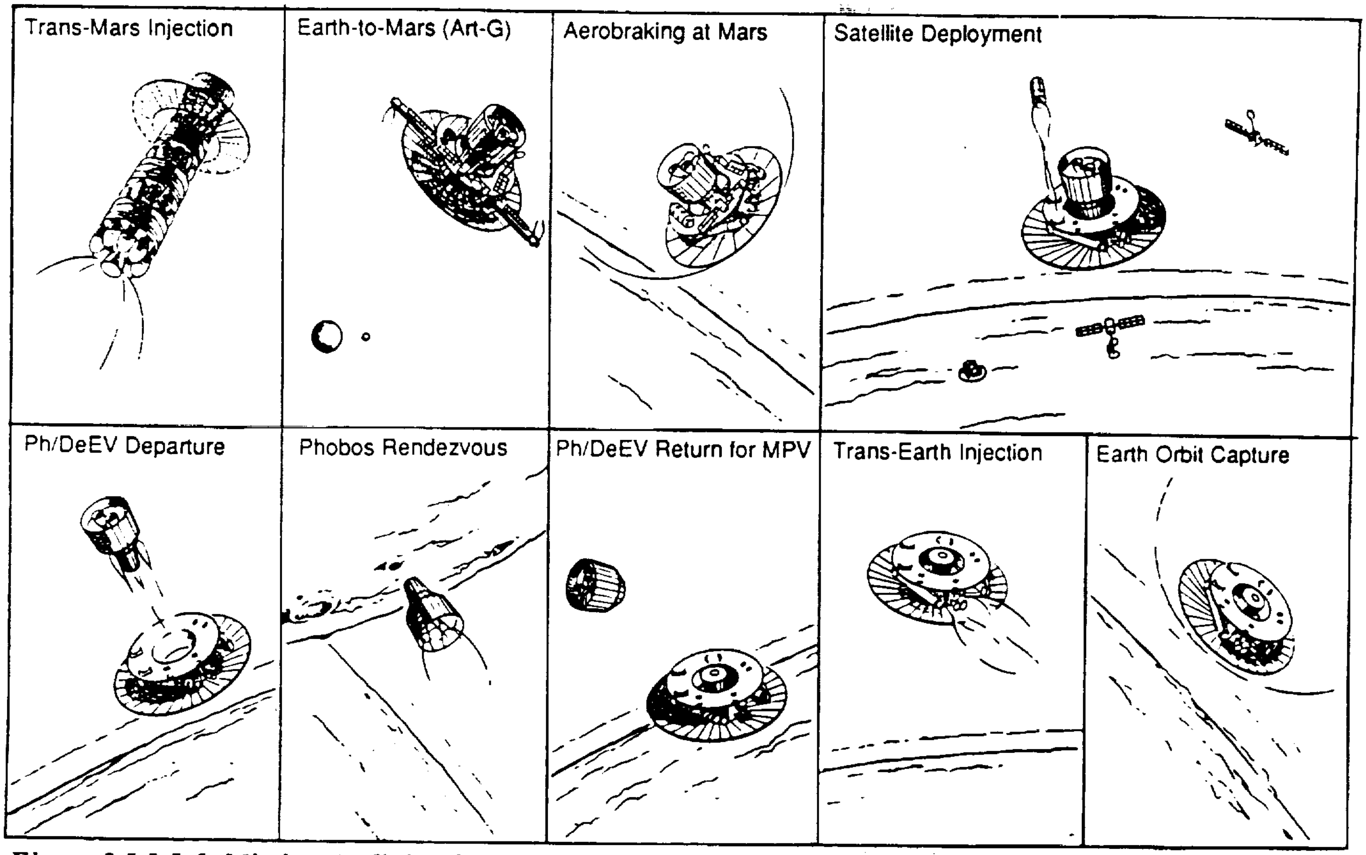


Figure 3.1.2.1-1 Mission-1 Flight Phases

13, 2006. Nominally they stay on the surface for 200 days, but could extend the stay to 500 days. For the 200 day stay they ascend to orbit and depart Mars on September 1, 2006. On November 13, 2008, after one and a half revolutions about the Sun they arrive at Earth. Again two aeropasses are required before rendezvousing with the assembly/refurbishment facility in LEO. This mission has an IMLEO of 687.2 tonnes and includes four 140 tonne TMI stages. However, one of these stages is only loaded with 3 tonnes of propellant. It also has a 184 tonne MPV that is identical to the first mission's but with less loaded TEI propellant. Finally, it carries a lander consisting of a crew cab, LAPM, AOTPM, and a landed habitation module. All of these excursion vehicle elements have a combined mass of 69 tonnes.

3.1.2.3 Mission-3: Gateway Development Cargo—The third mission has no crew but sends 150 tonnes of equipment to the selected gateway moon which was assumed to be Phobos, the larger moon. Earth departure occurs on October 5, 2007 and Phobos arrival, after a Mars aerocapture maneuver, occurs on November 25, 2008. Figure 3.1.2.3-1 shows the mission phases for this cargo flight. This mission delivers 50 tonnes of supplies destined for Mars which are attached to the necessary cargo lander plus an additional 25 tonnes of water mining, electrolysis and cryogenic liquefaction equipment, a 75 tonne dual-tether system, and finally, a 10 tonne vehicle changeout facility. The Mars surface payload is stored at Phobos until the next mission arrives. This is the heaviest flight with an IMLEO of 725.3 tonnes. It takes three full 140 tonne TMI stages and a fourth with 13 tonnes of loaded propellant. The Mars Cargo Vehicle (MCV) has a mass of 94 tonnes and the cargo lander (LAPM) has a mass of 27 tonnes.

3.1.2.4 Mission-4—The fourth mission is the first to use the facilities implanted at Phobos, hence, it does not need to leave Earth with TEI propellant or

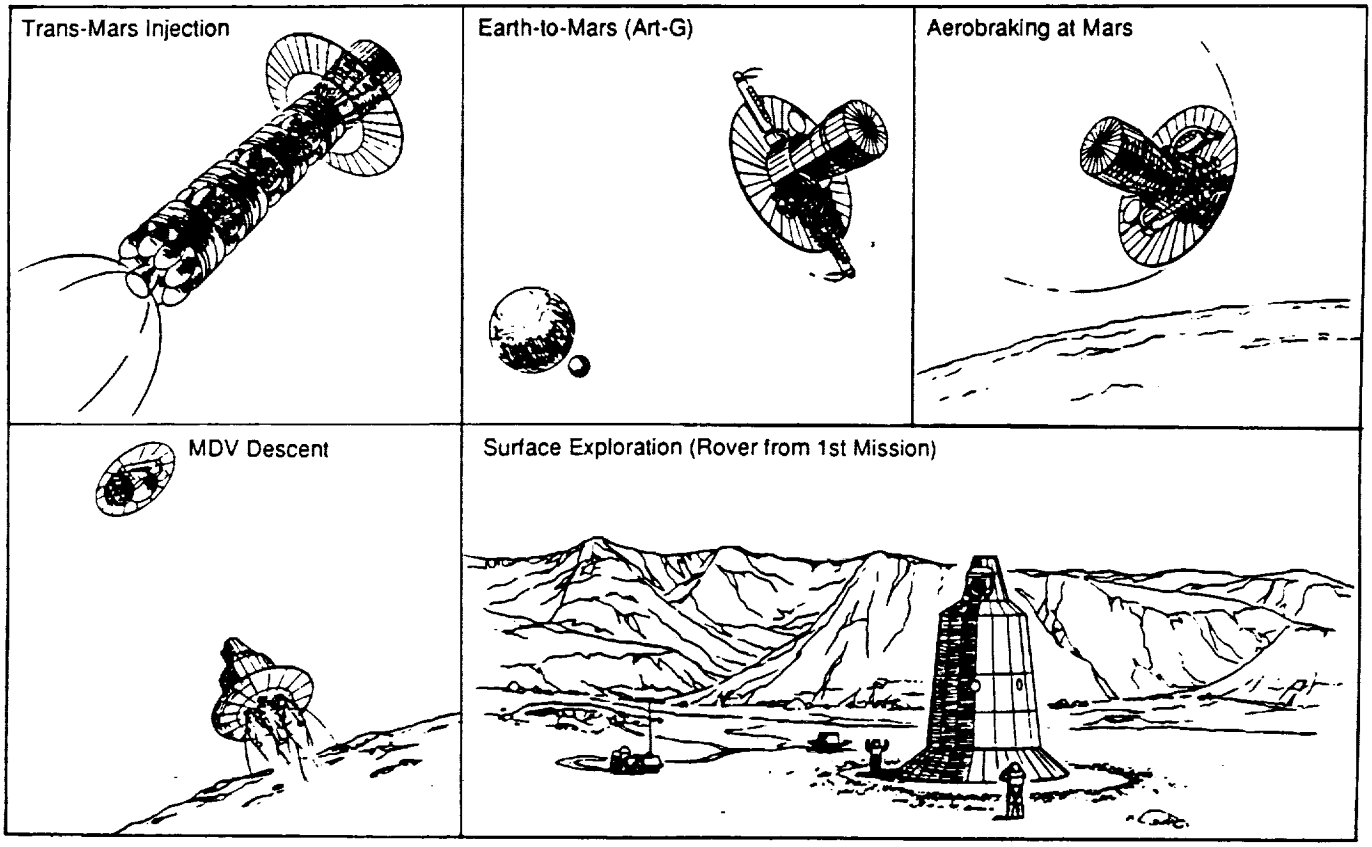


Figure 3.1.2.2-1 Mission-2 Flight Phases (Part One)

excursion vehicle propellant. The MPV, as shown in Figures 3.1.2.4-1 and 3.1.2.4-2, from mission-1 is used again and departs Earth on October 15, 2009 with seven crew aboard. It arrives at Phobos on November 27, 2010 after aerobraking at Mars. Once at Phobos the Mars Crew Sortie Vehicle (MCSV) is loaded with propellant and the crew descends to the small surface base on Mars. The crew spend nine months on the surface using the equipment and habitation modules landed on missions-2 and 3. They ascend in the MCSV to low Mars orbit where an AOTPM, recently dispatched from Phobos, docks with them and transfers them up to Phobos. On August 12, 2011 the MPV is attached to the tether and reeled out, away from Mars, and released throwing the MPV into a high energy orbit. After an apoapsis retro burn the MPV falls to within 250 km of Mars surface where it burns it's trans-Earth injection (TEI) engines to gain hyperbolic energy for Earth return. On July 10, 2012 the MPV performes a double aerocapture at Earth and returns to the assembly/refurbishment fixture. Because this mission used the tether facility and Phobos propellant, the mission's IMLEO drops to only 510 tonnes. This consists of two 140 tonne TMI stages, a third TMI stage with 23 tonnes of propellant, and a 169 tonne MPV that carries a 27 tonne MCSV that is not loaded with propellant.

3.1.2.5 Mission-5—Mission-5 repeats the sequence of mission-4 except that it uses the MCSV stored at Phobos from the previous mission. It departs Earth November 20, 2011 and arrives at Phobos June 25, 2012. The crew descends to the two habitation modules already on the surface in the MCSV. After 15 months on the surface the crew ascends back to Phobos, transfers to the MPV and, using tether assist, performs TEI on September 27, 2013. The MPV aerobrakes at Earth on August 28, 2014. Because Phobos propellant is used for the MCSV and for the MPV's Earth return propellant and because no excursion vehicle is taken to Mars, this is

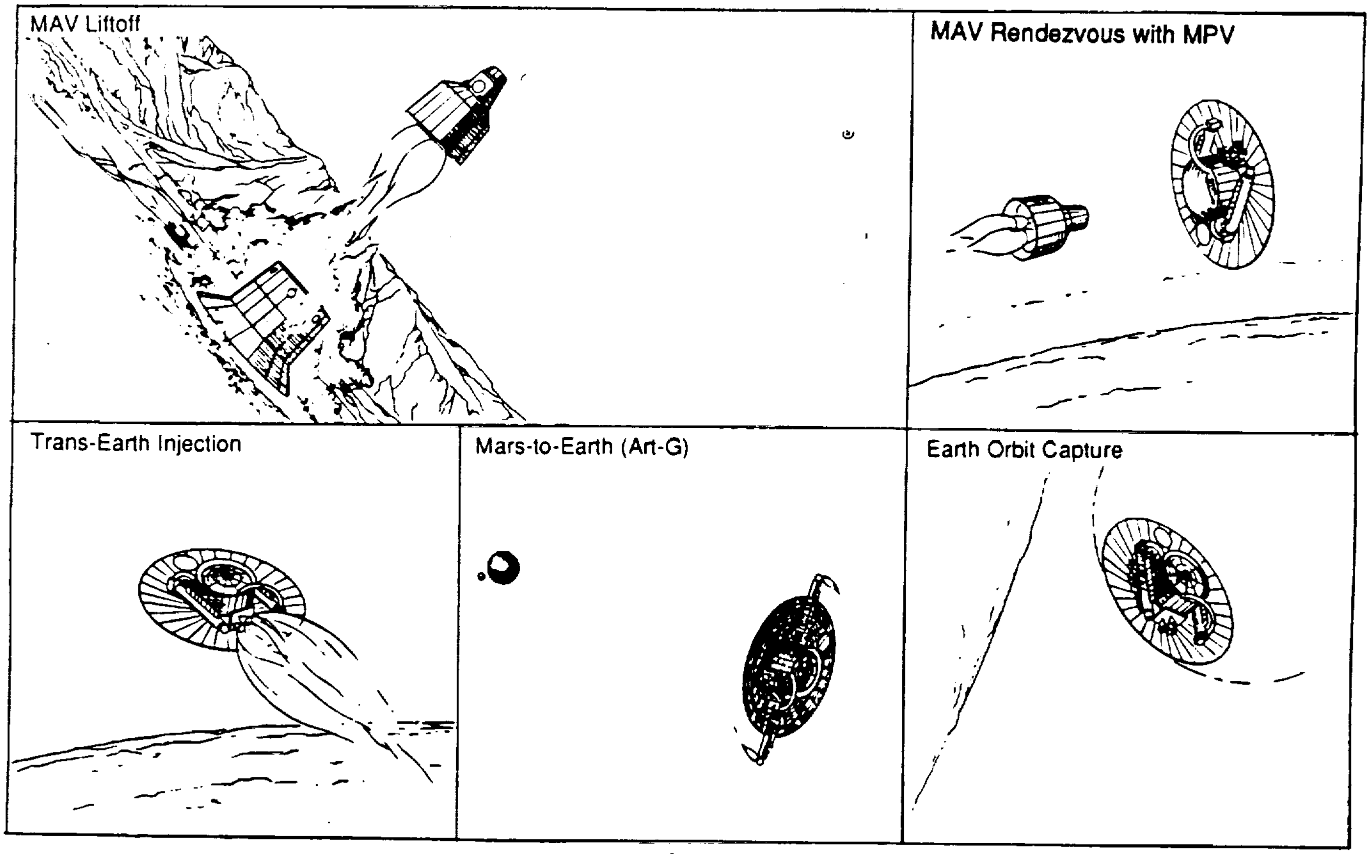


Figure 3.1.2.2-2 Mission-2 Flight Phases (Part Two)

the lightest mission with an IMLEO of only 428 tonnes. This breaks down into two TMI stages (one off-loaded by 21 tonnes) and a 170 tonne MPV.

3.1.2.6 Mission-6—Mission-6 begins the operational phase with the introduction of the first nuclear vehicle. The nuclear electric propulsion cargo vehicle (NEP-CV) has its heavy payload delivered to it from LEO by a chemically propelled space transfer vehicle (STV). From a 700 km nuclear safe orbit (NSO) the NEP-CV begins its spiral climb away from Earth. After 387 days it has escaped Earth's gravitational pull and begins to accelerate away from Earth on its way toward Mars. For all but two days out of 300 the NEP-CV's ion thrusters are firing as it climbs away from the Sun nearing Mars orbit. After 687 days it arrives at Mars and begins to spiral down toward Phobos, which takes another 96 days. Because of the high radiation outside of the reactor's shadow shield, the NEP-CV parks at the nuclear staging point, which is 1000 km behind Phobos in a chase orbit about Mars. From here the 400-tonne payload is picked up by the chemically propelled AOTPM and delivered to the Phobos facility. The NEP-CV then returns to NSO Earth in 411 days without a payload.

3.1.2.7 Mission-7—Mission-7 is the first use of a nuclear thermal rocket piloted vehicle (NTR-PV). As with the NEP-CV the NTR-PV departs from NSO, which means the crew of seven must transfer up to the 700 km basing altitude in an ECCV/AOTPM. Transfer to Mars takes only 126 days for the March, 2015 opportunity. After aerobraking at Mars the NTR-PV circularizes at a point 1000 km behind Phobos in the same nuclear vehicle staging zone used by the NEP-CV. An MCSV goes out, retrieves the crew, and takes them directly to the Mars surface base. After two years the MCSV lifts off and returns the crew to the NTR-PV. After a 139-day transfer back to Earth the NTR-PV performs a double aerocapture at Earth and parks at

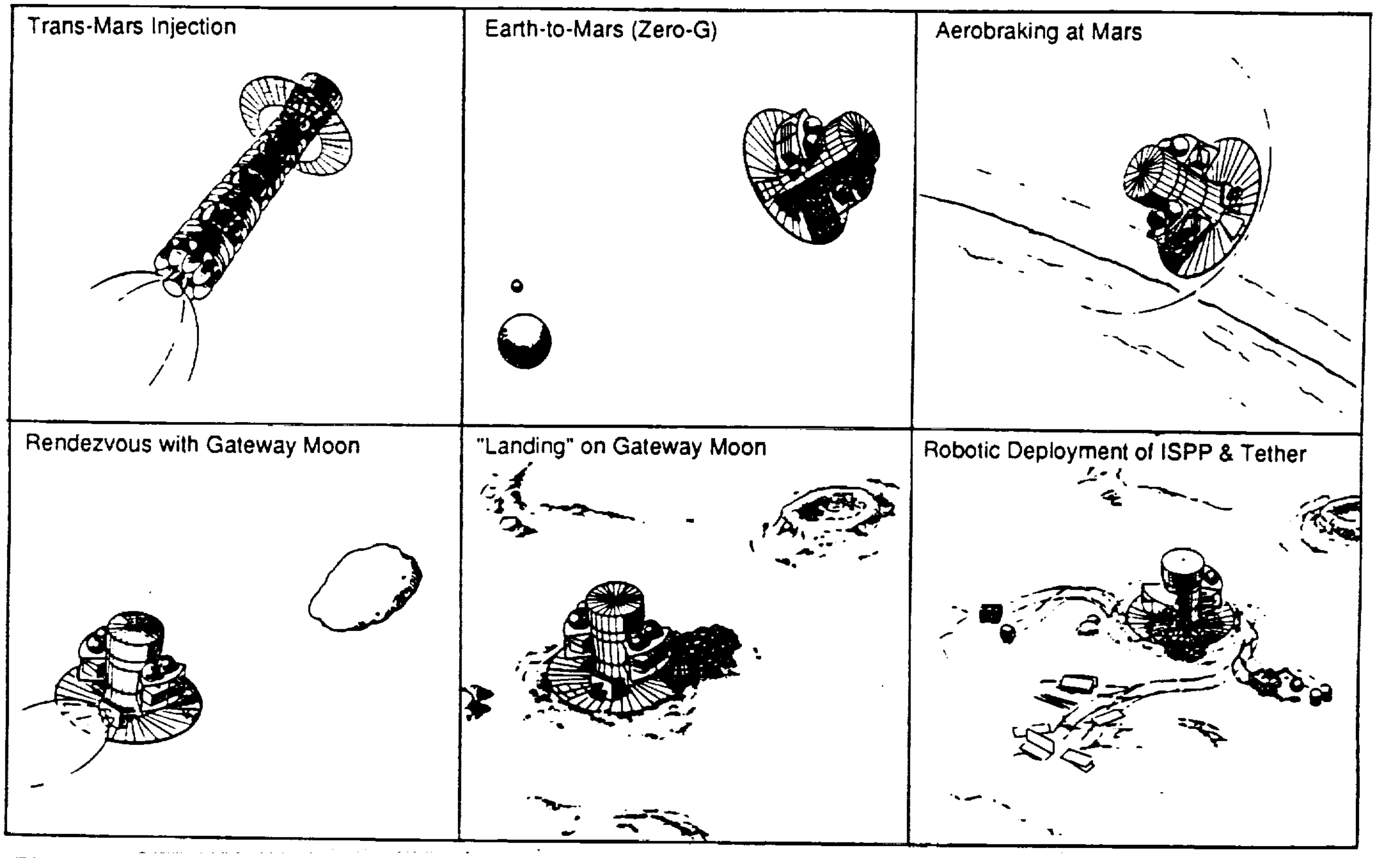


Figure 3.1.2.3-1 Mission-3 Flight Phases

NSO. Finally, an ECCV/AOTPM returns the crew to the space station.

3.1.3 Requirements

From the program objectives stem a set of requirements defining implementation boundaries for vehicles and technologies. The most significant of these are summarized below. To start with, the Mars Piloted Vehicle is to be chemically propelled using hydrogen and oxygen propellants. Additionally, it will use aerobrakes at Mars and Earth and it will be recovered and reused at Earth. Further, it will carry up to seven crew and must provide safe haven and dual egress capabilities from all habitable sections. Also, it will provide artificial gravity of at least one third Earth's with a spin rate not to exceed 4 rpm. Finally, propellant tanks must be sized sufficiently large to enable a post-TMI ΔV capability of 2.86 km/s.

Excursion vehicles will be expendable with a cargo lander able to land 50 tonnes of equipment and a personnel lander able to take seven crew and 10 tonnes from Phobos to Mars' surface and back again. It should be noted that during the first cycle of studies a single-stage fully reusable Mars lander/ascent vehicle was specified for crew transfer, but it was extremely massive because it had to perfrom 7 km/s ΔV. During the second cycle the requirement was dropped but the effort continued to develop a fully reusable vehicle. By using an orbital space tug the reusable single-stage MCSV became efficient. Hence, a reusable MCSV, although not required is present in the final scenario.

The requirement to make vehicle configurations common led to the concept of modules for the excursion vehicles and to common MPV and Mars Cargo Vehicle aerobrakes, structures, and propulsion systems.

It was also mandated that tethers be used at Phobos for propellant handling and vehicle momentum exchange to reduce propellant needs. This influenced both excursion and interplanetary vehicle design and implementation.

The final set of significant requirements center around Earth-to-orbit (ETO) vehicles. ETO launch limitations require that payloads fit in a 12.5 x 25-meter cylindrical envelope, that no more than 570 tonnes be launched per year (90 tonnes of which is hardware), and that launches be spaced by at least 45 days with no more than four launches per year. Launch destination is to be a 500-km circular orbit inclined at 28.5 degree which is Space Station Freedom's orbit.

3.1.4 Assumptions

Several assumptions have to be made before vehicles can be designed. First, it is assumed that Phobos will be the gateway moon. Additionally, on Phobos, a single tether (with a backup) will be used for both upward (away from Mars) and downward (toward Mars) deployments. Also, the MCSV is assumed to be made reusable by introduction of the Mars orbital space tug (an AOTPM) and the Phobos tether system. Further, all vehicle cabin pressures will be at sea level pressure (100 kPa). And finally, the heavy-lift launch vehicle (HLLV) has a 10-meter useable inside diameter and it uses a 140 tonne upper stage which is made common with the trans-Mars injection stage.

3.2 VEHICLE DESCRIPTIONS

Nine vehicles are designed for Mars Evolution: four interplanetary transfer ships, four excursion vehicles and one rescue capsule. The excursion vehicles and rescue capsule are all made up of standardized modules for reduced cost. Below are definitions of these vehicles and modules. After

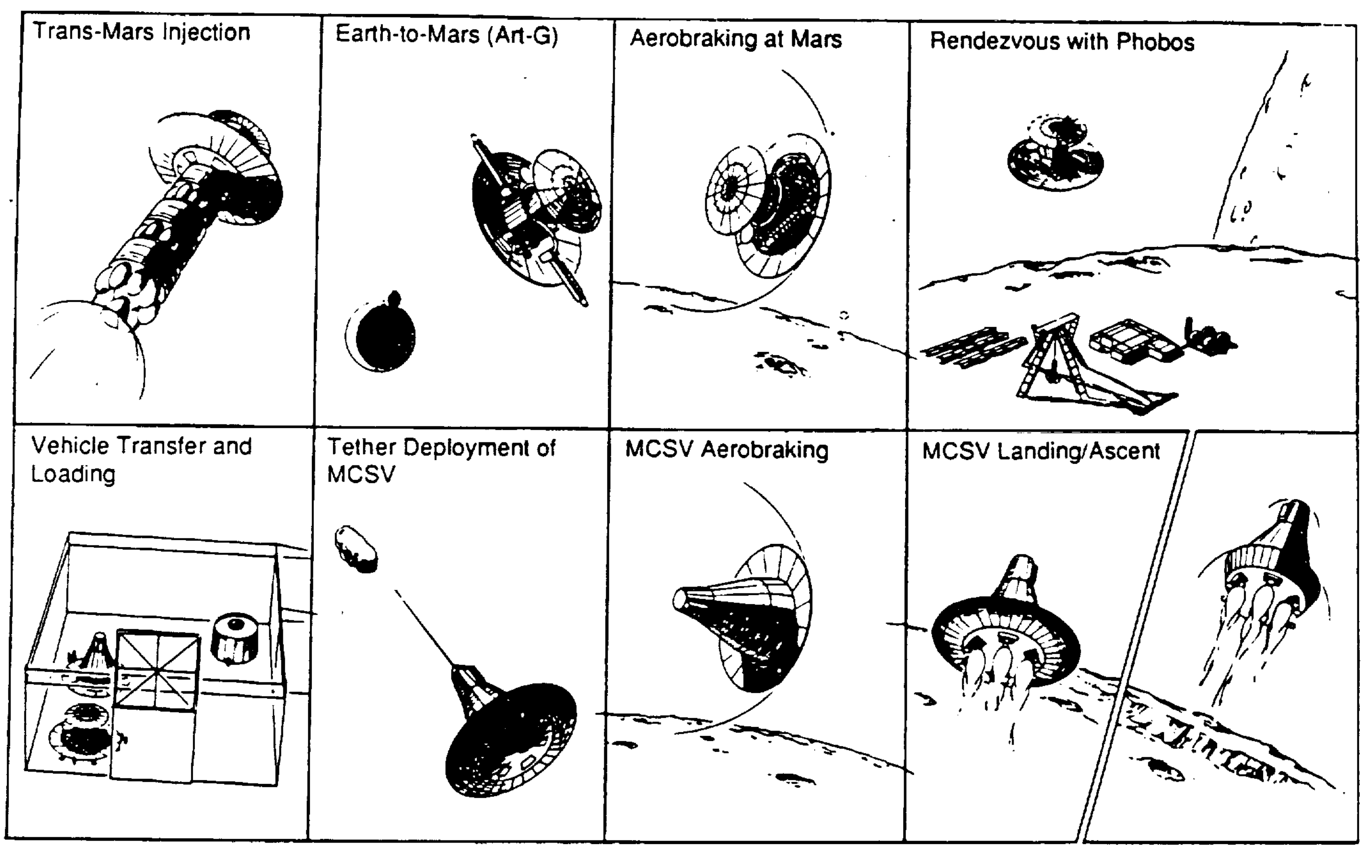


Figure 3.1.2.4-1 Mission-4 Flight Phases (Part One)

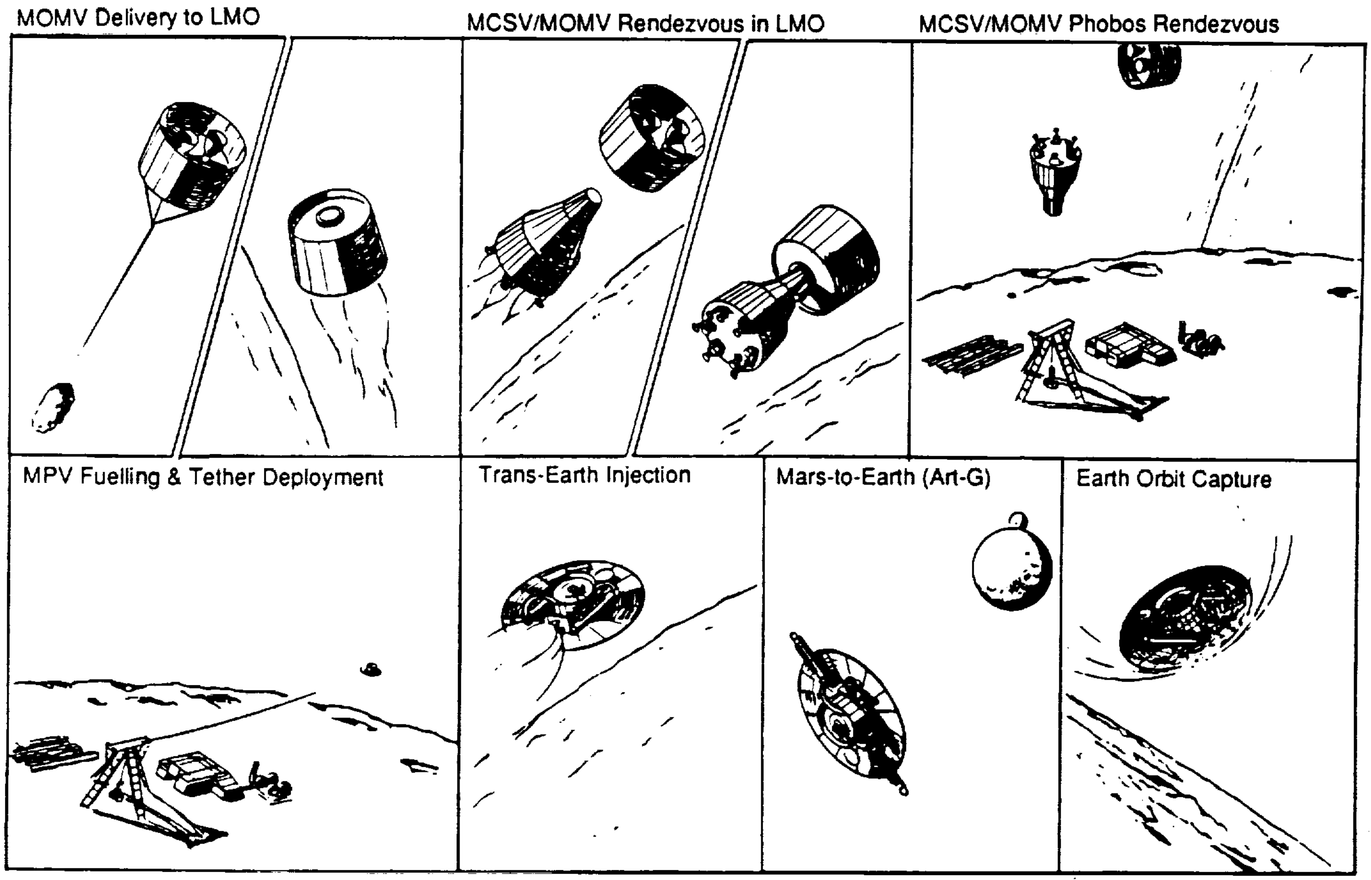


Figure 3.1.2.4-2 Mission-4 Flight Phases (Part Two)

each vehicle's description is a definition of the modules that make up that vehicle–unless they have already been defined in a previous section.

3.2.1 Configurations

3.2.1.1 Mars Piloted Vehicle—The MPV carries 3-7 crew from LEO to Mars orbit and back again to LEO. Figure 3.2.1.1-1 gives the dimensions of the vehicle in meters. The structural backbone of the vehicle, as shown in Figures 3.2.1.1-2 and 3.2.1.1-3, is a set of three 10-meter rings at the center of an umbrella-like aerobrake that unfolds to a 39-meter diameter after being launched into LEO. In the center of these rings are the integral TEI propellant tanks. Six cylindrical tanks between the center and upper rings holds oxygen and a single cylindrical tank between the center and lower rings holds hydrogen. Although the volume of these tanks could carry 128.6 tonnes of propellant, most missions use significantly less. Three advanced STV engines are mounted outside the rings with a total thrust of 100,000 Newtons and an Isp of 480 seconds. In-between the oxygen tanks is the central docking hub where the crew and supplies are loaded and unloaded. The hub connects to the two space-station sized habitation modules with two tunnels, one on each side. Each tunnel attaches to one end of a habitation module through a pressurized bearing that allows full access to each module, the central docking hub, and any excursion vehicle or rescue capsule docked to the docking hub. Four swing-out attachment fittings are mounted on the top ring that structurally supports an excursion vehicle, cargo lander, or any generic cargo designed to interface with the fittings.

For all aerobraking and large propulsive maneuvers the habitation modules are in their stowed position, attached at each end, within the 30-degree wake

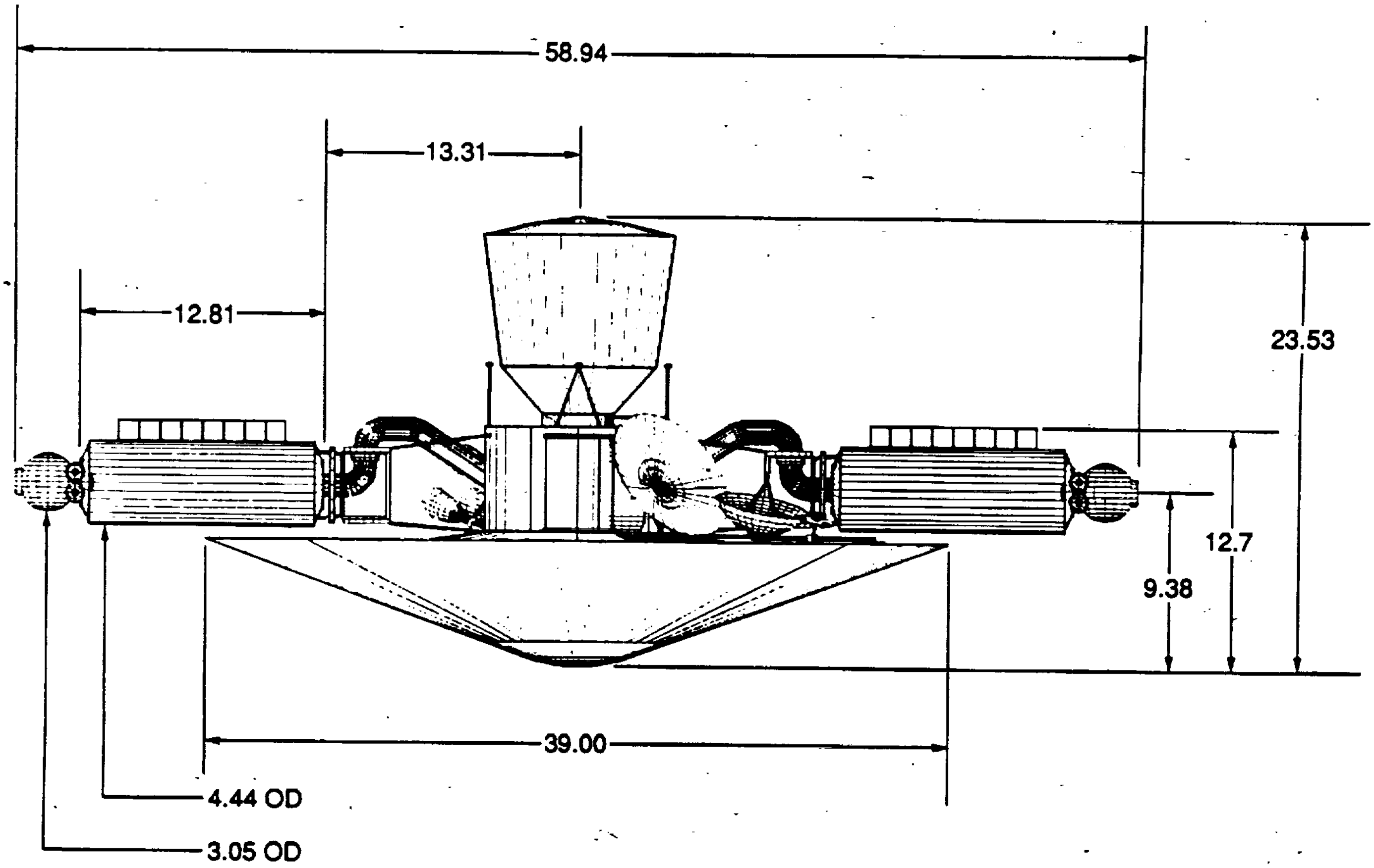


Figure 3.2.1.1-1 Mars Piloted Vehicle with Dimensions (in Meters)

cone that extends up from the aerobrake's rim during aerocapture. For interplanetary cruise and while parked in Mars orbit, the habitation modules are rotated out, propeller fashion, to increase their mean distance from the center of the vehicle, thereby increasing the mean centripetal acceleration felt by the crew.

Opposite the TEI engines is a cantilever truss that supports several communications antennas including the 5-meter high-gain antenna that provides at least a 10 Mbps data rate to Earth. A small boom swings out beyond the aerobrake rim to provide hemispherical antenna and science instrument views "behind" the aerobrake. The inner structure of this truss supports consumables and science experiments that do not require crew access.

Trans-Mars injection (TMI) is accomplished with several Shuttle-Z upper stages, each with a loaded mass of 140 tonnes and a dry mass of 13 tonnes, as shown in Figure 3.2.1.1-4. Each stage is 10-meters in diameter and 11-meters long and consists of one cylindrical hydrogen tank, eight stretched sphere oxygen tanks, and a single high expansion ratio SSME engines nested inside the oxygen tanks. Because these stages arrive in LEO with depleted

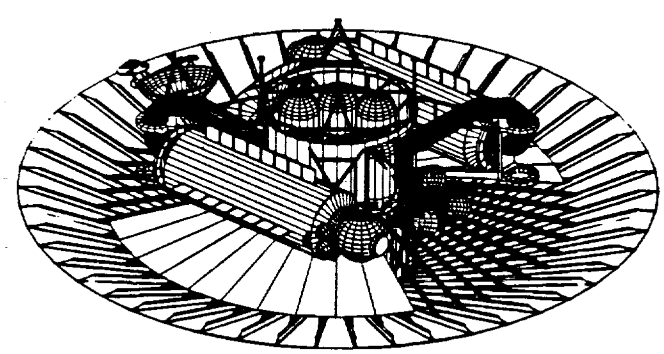


Figure 3.2.1.1-2 Mars Piloted Vehicle–Zero-G Configuration

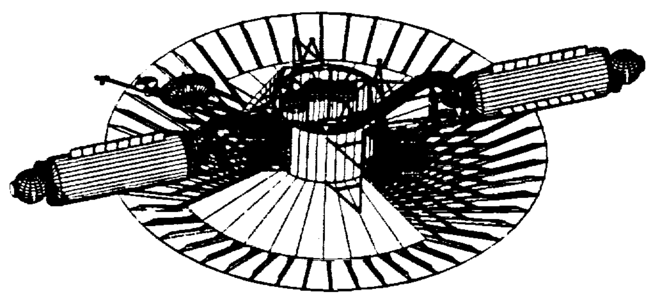


Figure 3.2.1.1-3 Mars Piloted Vehicle–Artificial Gravity Configuration

propellant loads they must be refueled once in orbit. This occurs after they have been integrated together and attached to the MPV, just before TMI.

3.2.1.2 Mars Cargo Vehicle (MCV)—The MCV of Figures 3.2.1.2-1 and 3.2.1.2-2 carries up to 187 tonnes of cargo from LEO to Phobos and has several elements common with the MPV. The aerobrake is identical and the TEI propulsion system is the same except that only four of the six oxygen tanks are installed. Although the MCV does not return to Earth it still needs a significant post-TMI propulsion capability for trajectory correction maneuvers and rendezvous with Phobos after aerobraking at Mars. Payload is accommodated on double-deck platforms that cantilever out from the central ring structure and provide 365 m^2 of attachment surface. Payloads and science instruments are given the same accommodations as on the MPV.

TMI is achieved with the same stages used for the MPV. In the reference cargo mission it takes four of these stages to give the MCV the required velocity out of LEO.

3.2.1.3 Nuclear Electric Propulsion Cargo Vehicle (NEP-CV)—The NEP-CV is designed to carry 400 tonnes from NSO (700 km) about Earth to a staging point near Phobos and return empty. It uses a 26.7 megawatt thermal reactor capable of generating five megawatts of electric output using a closed loop Brayton cycle. The radiator is conical shaped, and is located just inside the radiation shadow created by a disk-shaped shield. It has a radiating surface of 3694 m^2. Figure 3.2.1.3-1 shows the vehicle, payload, and science instrument accommodation provided along the central truss/

Dry Mass (includes residuals)	13000 kg
Payload Mass	Variable
Propulsion System	
Propellant Type	Chemical-LH_2/LOX
Engines	
Number	One
Type	SSME/HER
Mass (ea.)	3628 kg
Thrust (total)	2372 kN
I_{sp} (471 sec)	4623 m/s
Nozzle Diameter	4.5 m
Propellant Mass	127,000 kg
Tank Mass	8890 kg
Initial T/W_E (no payload	1.7
Mass Fraction (no payload)	0.907
Total Length (nested nozzle)	11 m
Total Mass (without payload)	140,000 kg

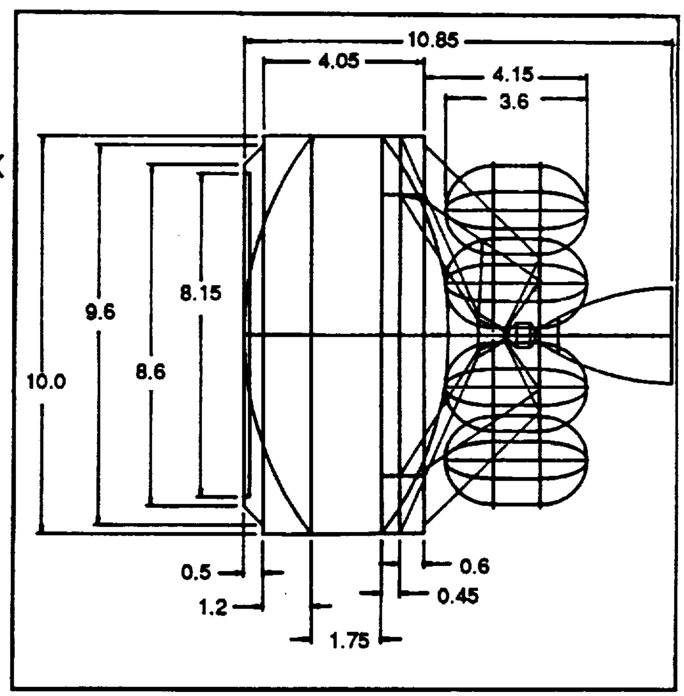


Figure 3.2.1.1-4 Trans-mars Injection Stage/Shuttle-Z Upper Stage (Dimensions are in Meters)

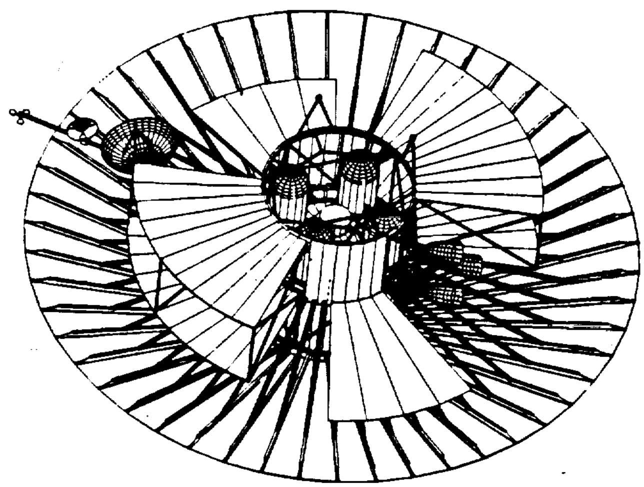


Figure 3.2.1.2-1 Mars Cargo Vehicle

spine. Propulsion is provided by 16 ion thruster operating at an Isp of 6000 seconds. Loaded vehicle mass, without payload, is 276 tonnes, which includes 167.6 tonnes of argon. Transfer time from NSO to Phobos with a 400-tonne payload takes 783 days. Because vehicle acceleration is inversely related to payload mass, longer or shorter trip times result from heavier or lighter payloads respectively.

3.2.1.4 Nuclear Thermal Rocket Piloted Vehicle (NTR-PV)—The NTR-PV delivers seven crewmembers from NSO about Earth to Phobos and back again on high energy trajectories taking between 104 and 173 days for opportunities beginning in 2011. This compares to 220-300 days for minimum energy trajectories used chemically propelled vehicles. The backbone of the vehicle is an 81-meter long by 12.5-meter wide aeroshell as seen in Figure 3.2.1.4-1. Loaded into this structure are two interconnected 2/3 length space station modules, two tandem 140 tonne hydrogen propellant tanks with tapered bottoms, and the reactor and its shielding. Total loaded mass is 355.8 tonnes, 225 tonnes of which is propellant. By rotating the ship at 4 rpm a sensed gravity of 0.6 times that of Earth's can be generated. The reactor creates 3000 megawatts of thermal energy that when imparted to the hydrogen flow gives Isp of 900 seconds and thrust of 668 kN.

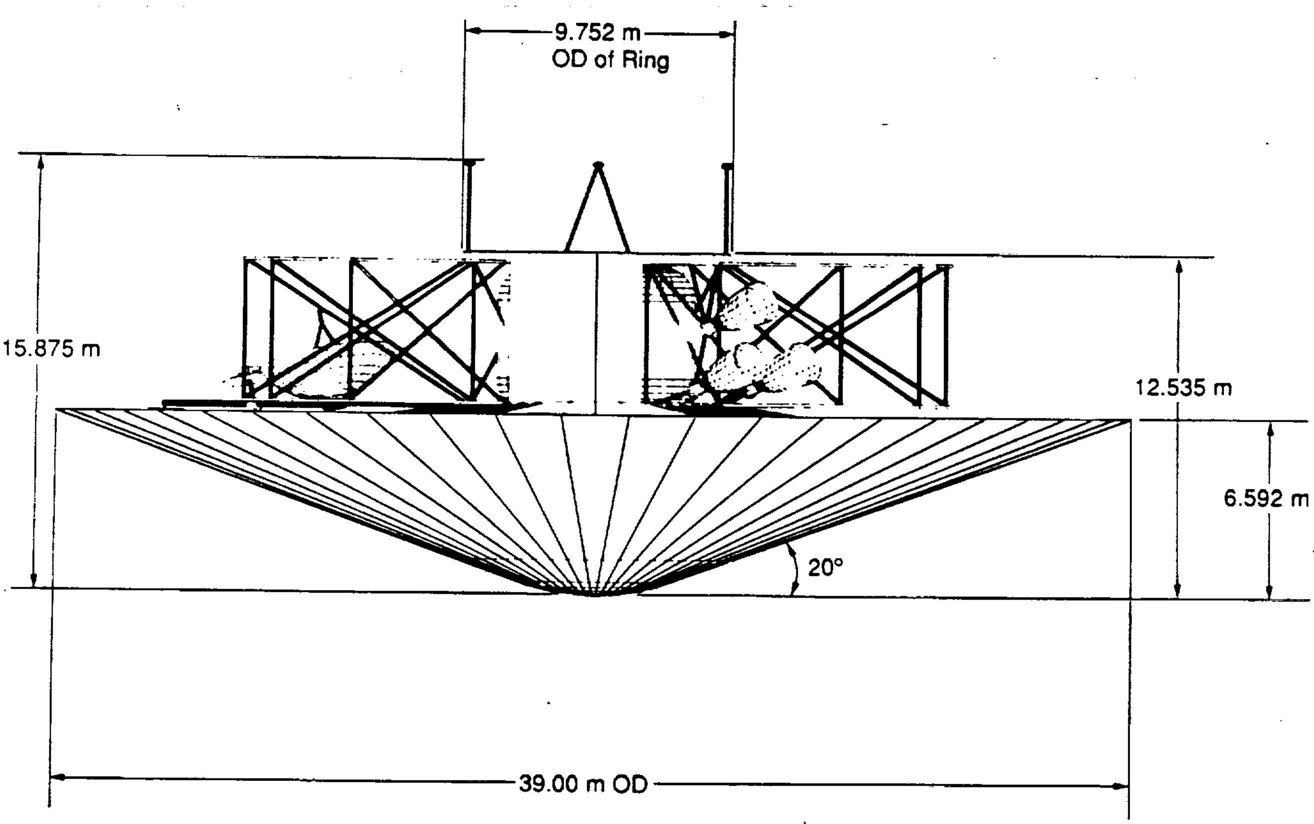


Figure 3.2.1.2-2 Mars Cargo Vehicle with Dimensions

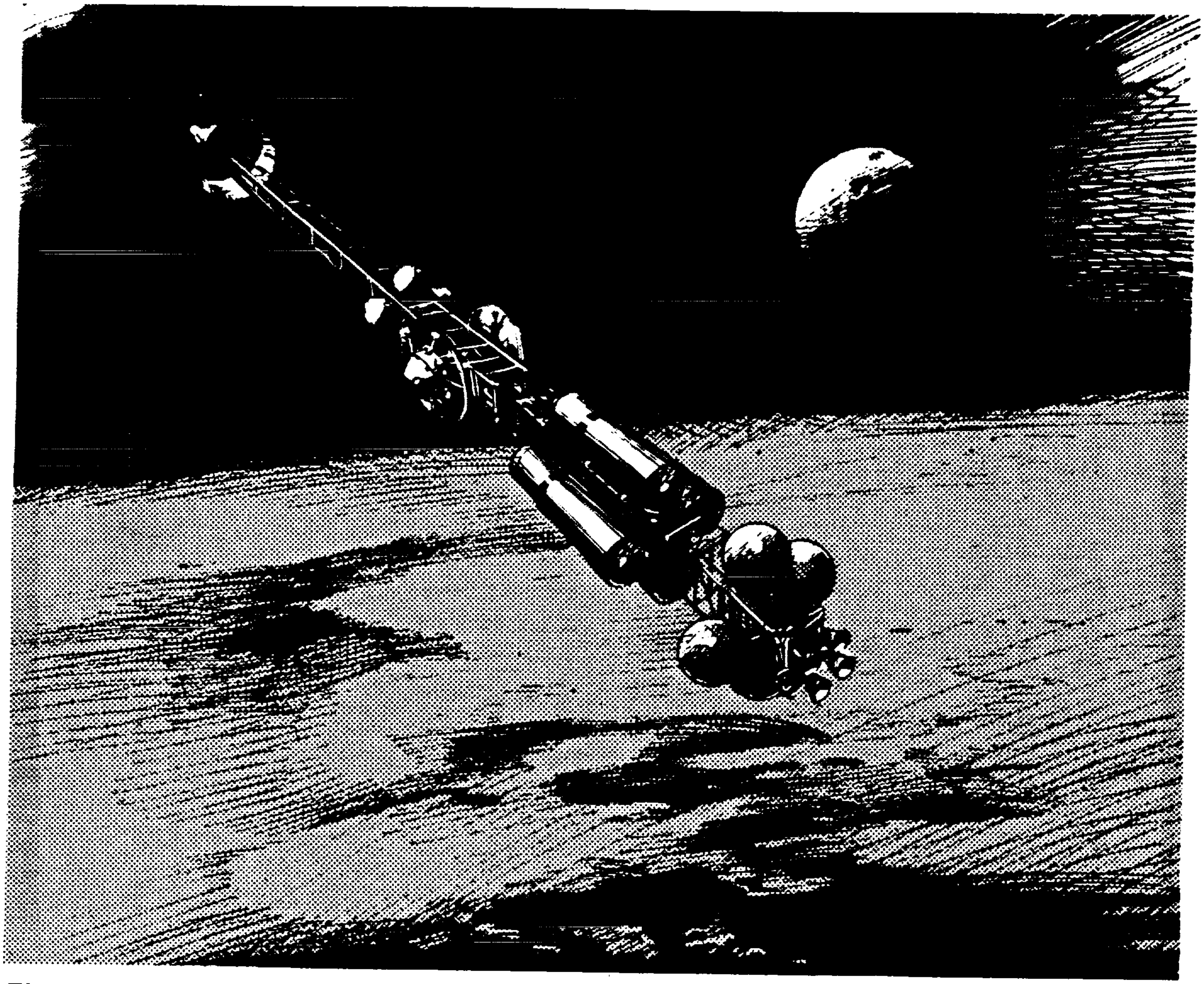


Figure 3.2.1.3-1 Nuclear Electric Propulsion–Cargo Vehicle

3.2.1.5 Excursion Vehicles—The Mars Evolution case study uses four excursion vehicles and a rescue capsule which is called the ECCV. The three excursion vehicles are the Mars Descent Vehicle (MDV), Phobos and Deimos Excursion Vehicle (Ph/DeEV), Mars Crew Sortie Vehicle (MCSV), and the Mars Cargo Lander (MCL). To reduce program costs all of these vehicles are made up of shared modules. These modules are the Crew Cab Module (CCM), Lander/Aerobraking Propulsion Module (LAPM), Ascent and Orbit Transfer Propulsion Module (AOTPM), and Ascent Propellant Module (APM). Figure 3.2.1.5-1 shows how different excursion vehicles are created with modules.

3.2.1.5.1 Phobos/Deimos Excursion Vehicle—The Ph/DeEV carries two crew from the MPV, parked in a 250 x 18000-km orbit, to Phobos then to Deimos and back again to the MPV. It is carried to Mars on the MPV attached to the four struts on the uppermost ring frame and is accessed through the central docking hub. Two modules constitute the Ph/DeEV: the CCM and AOTPM. Figure 3.2.1.5.1-1 gives a see-through side view of the Ph/DeEV. Dimensions for this and all other vehicles are in meters.

3.2.1.5.1.1 Crew Cab Module—The CCM is designed to carry up to seven crew for intervals less

Figure 3.2.1.4-1 Nuclear Thermal Rocket–Piloted Vehicle

than 2 days. Figure 3.2.1.5.1.1-1 give the specifics and several views of the CCM. It has a flight deck for the commander and pilot and a passenger deck with five couches in a row. Beneath the passenger deck is a service deck that contains crew and vehicle consumables, storable propellants for the module's reaction control system (RCS), thrusters, and attach points and interfaces for other modules. The RCS system is designed to an equivalent capability of 500 meters per second of linear ΔV.

3.2.1.5.1.2 Ascent and Orbit Transfer Propulsion Module—The AOTPM is designed to propel the CCM either from orbit-to-orbit or off Mars' surface into orbit. It also becomes a Phobos based reusable upper stage that retrieves the MCSV from low Mars orbit (LMO) and acts as a backup to the Phobos tether momentum transfer system. It consists of two RL-10B-2 engines, two hydrogen tanks, and two oxygen tanks surrounded by a shroud/meteor shield. The crew cab module controls the AOTPM except in it's Phobos upper stage role, in which case an avionics package is added that provides a flight computer, inertial reference units, star trackers, and telemetry capabilities. Figure 3.2.1.5.1.2-1 gives the specifics of the design and shows the tank and engine layout.

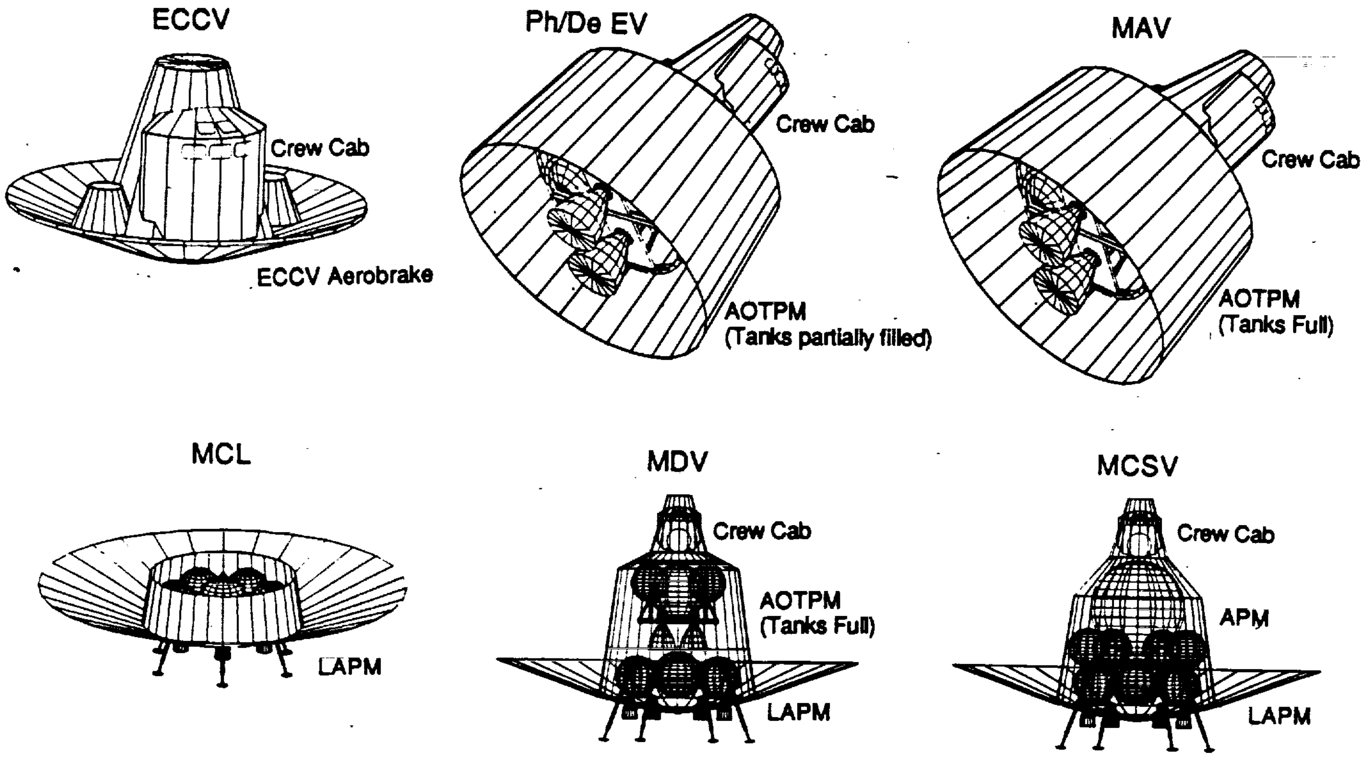


Figure 3.2.1.5-1 Summary of Excursion Vehicle and Their Modules

3.2.1.5.2 Mars Descent Vehicle—The MDV descends to the surface of Mars from a 250 x 33120-km orbit, leaves the lander/aerobrake propulsion module there and, after an extended surface stay, ascends back into a high energy orbit. It consists of three modules: CCM, AOTPM, and LAPM and is shown in Figure 3.2.1.5.2-1. The first two modules are defined above. The LAPM is the only new module that enables the Ph/DeEV to become an MDV.

3.2.1.5.2.1 Lander/Aerobrake and Propulsion Module—The LAPM is designed to carry up to 50 tonnes from high Mars orbit to the surface. In it's initial role it remains on the surface, but later on, it folds in its aerobrake skirt, fires its engines and ascends back to orbit using propellant from the Ascent Propellant Module. Figure 3.2.1.5.2.1-1 shows the LAPM with its brake deployed and gives the specifications of the design. The LAPM consists of a tapered disk main body that is 9.91-meters wide at the bottom. To protect the base during

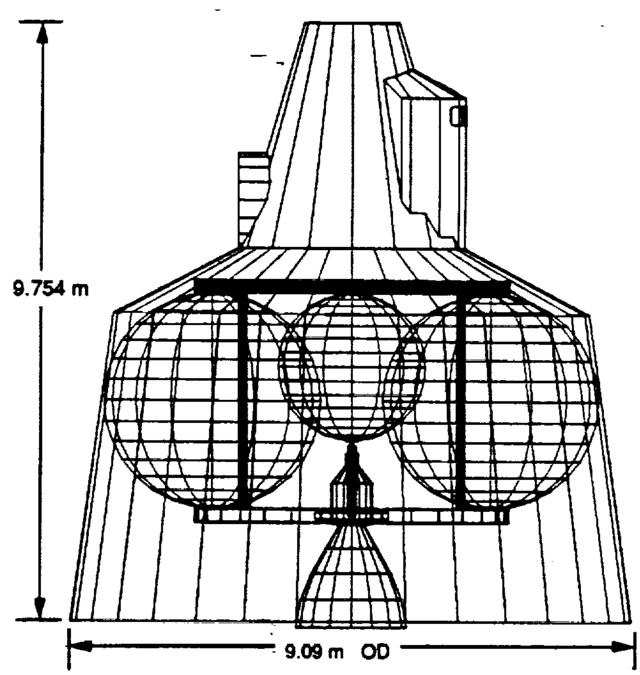


Figure 3.2.1.5.1-1 Phobos/Deimos Excursion Vehicle

Dry Mass	
(includes 7crew, suits, consum, P/L)	4504 kg
Payload Mass (above items+200kg)	1670 kg
Payload Volume	m^3
Propulsion System	
Propellant Type	Storable Bi-Prop
Engines	
Number	30
Type	Marquart R-4D
Mass (ea.)	3.76 kg
Thrust (total)	444.8 N
I_{sp} (316 sec)	3050 m/s
Propellant Mass	805 kg
Tank Mass	40 kg
Initial T/W_M	0.01
Mass Fraction	0.15
Cabin Pressure (14.7 psia)	100,000 Pa
Total Mass (including all Payload items)	5309 kg

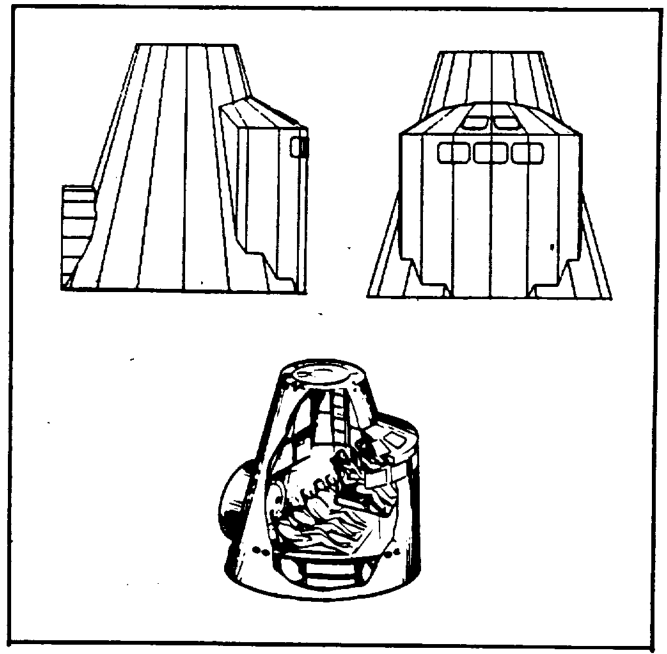


Figure 3.2.1.5.1.1-1 Seven Person Crew Cab Module

entry, the bottom has FRCI-20-12 ceramic tiles with six combined leg and engine doors around its perimeter. An additional flexible fabric skirt attached to the base decreases the ballistic coefficient and hence the peak heating rate. This skirt extends the diameter to 23 meters when deployed. Inside the taper disk core are six RS-44 class engines with 225:1 expansion ratios operating at a thrust of 66700 Newtons each and at an Isp of 463 seconds. Next to each engine is a pneumatic telescoping leg. Interspersed between the engines are six spherical hydrogen tanks surrounding a single spherical

Dry Mass	
(includes residuals)	2584 kg
Payload Mass (LAM/APM/Cab)	25847 kg
Payload Volume	Unrestricted
Propulsion System	
Propellant Type	Chemical-LH_2/LOx
Engines	
Number	2
Type	RL-10B-2
Mass (ea.)	191 kg
Thrust (total)	97.86 kN (22 klbf)
I_{sp} (460 sec)	4511 m/s
Propellant Mass	12648 kg (maximum)
Tank Mass	1264.8 kg
Initial T/W_E	0.75
Mass Fraction (Without Payload)	0.8303
Total Mass (Without Payload)	15232 kg

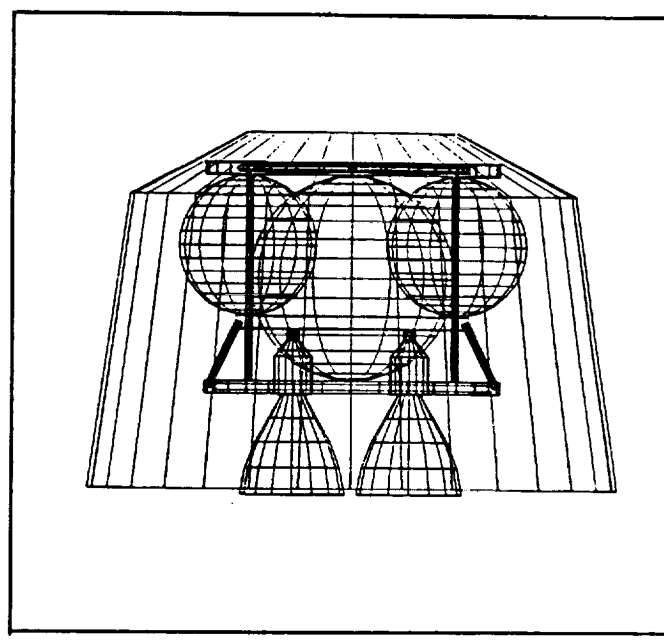


Figure 3.2.1.5.1.2-1 Ascent and Orbit Transfer Propulsion Module

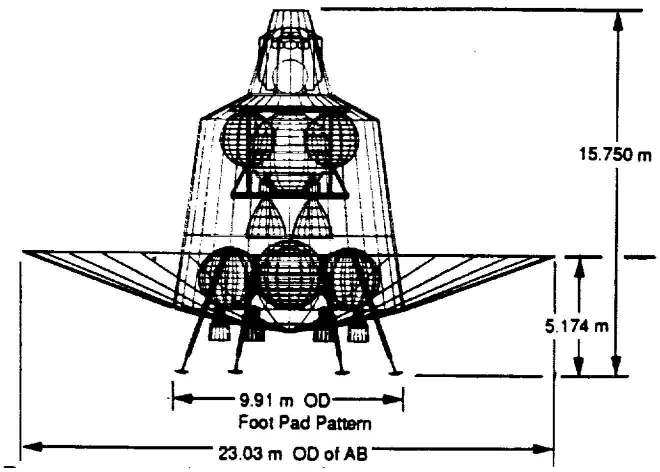


Figure 3.2.1.5.2-1 Mars Descent Vehicle with Dimensions

oxygen tank. When the LAPM is used in the MCSV it must fold in the aeroskirt for ascent. To keep air from forcing the folded skirt open, a protective rim causes the air to flow around the pleats of the folded brake material. All avionics and sensors required for landing are located on the inside of the tapered cylinder wall.

3.2.1.5.3 Mars Crew Sortie Vehicle—The MCSV as illustrated in Figure 3.2.1.5.3-1 carries up to seven crew from Phobos to Mars' surface, waits for over a year on the surface, and then ascends into a low orbit about Mars where it rendezvous with the the Phobos based upper stage that carries it back to Phobos. Unlike the MDV, the MCSV does not leave its lander on the surface. To bring it back to orbit a new module is required called the Ascent Propellant Module (APM).

3.2.1.5.3.1 Ascent Propellant Module—The APM carries hydrogen and oxygen to feed the LAPM's engines. Figure 3.2.1.5.3.1-1 shows a bottom view and gives the design values of the APM. It consists of a single spherical hydrogen tank surrounded at its base by six spherical oxygen tanks all inside a shroud/meteor shield. Propellant feed and autogenous gas lines at its base attach to the LAPM lines and feed the engines directly. The central hydrogen tank has a combination spray-on foam and multi-layer insulation system combined with a reliquefaction system to minimize heat flow into the cryogenic propellants both on Mars' surface and in space. Reliquefication pumps in the APM prevent propellant loss and are powered by fuel cells during flight and by surface based power when landed.

3.2.1.5.4 Mars Cargo Lander—The MCL delivers 50 tonnes of cargo from Phobos to Mars' surface. It

Dry Mass (includes residuals)	11,604 kg
Payload Mass (SRD Requirement)	50,000 kg
Payload Volume	360 m^3
Propulsion System	
Propellant Type	Chemical-H/O
Engines	
Number	6
Type	RS-44 class (225:1)
Mass (ea.)	155 kg
Thrust (total)	66.7 kN (15 klbf)
I_{sp} (463 sec)	4545 m/s
Propellant Mass (H/O only)	11,847 kg (maximum)
Tank Mass (H/O only)	376 kg
Initial landing T/W_M	1.6
Take-off T/W_M	1.76
Mass Fraction (With 50t Payload)	0.161
Total Mass	23,451 kg

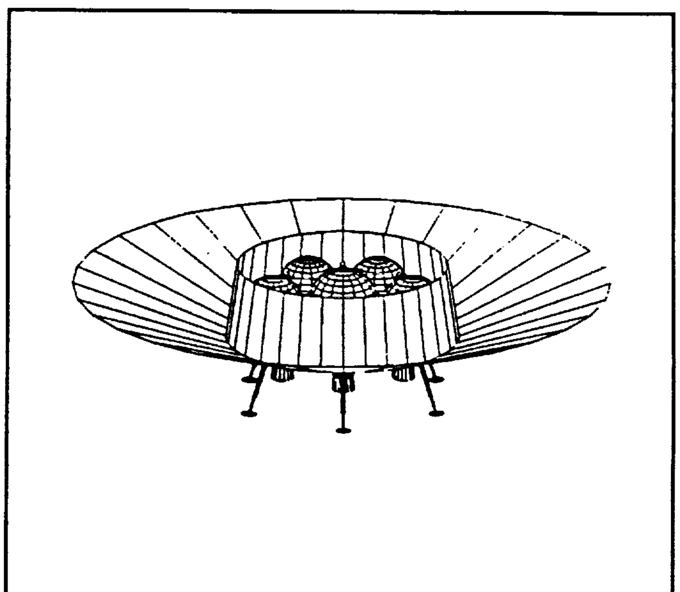


Figure 3.2.1.5.2.1-1 Lander/Aerobrake Propulsion Module

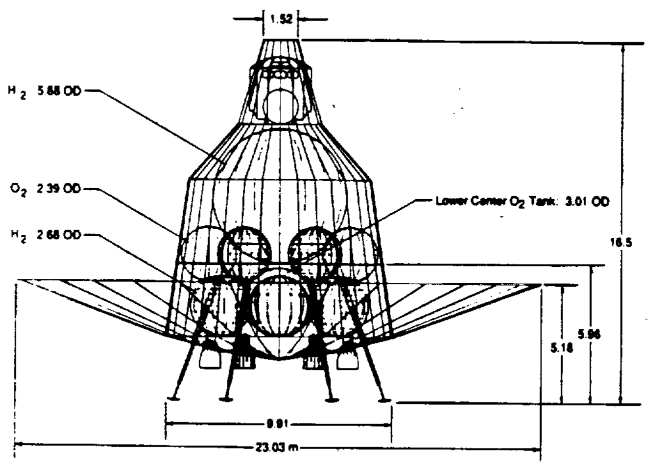


Figure 3.2.1.5.3-1 Mars Crew Sortie Vehicle

is one and the same as the LAPM seen in Figure 3.2.1.5.2.1-1 and used for the MDV and MCSV, but with an added avionics package for guidance and sequencing. It is not reusable and is carried to Phobos by either a MPV or a MCV.

3.2.1.5.5 Earth Crew Capture Vehicle—The ECCV provides a redundant means of returning the crew of an MPV to Earth. The ECCV consists of a crew cab module and a small rigid aerobrake as shown in Figure 3.2.1.5.5-1. The ECCV rides from Mars to Earth docked to the central docking hub of the

Dry Mass (includes residuals) (APM only, no payload or other Modules)	3153 kg
Payload Mass (no crew cab attached)	10482 kg
Payload Mass (with loaded crew cab)	5173 kg
Propulsion System	
Propellant Type	Chemical-H/O
Engines	
Number	None (Uses LAPM's)
Type	
Mass	
Thrust	
I_{sp}	
Propellant Mass	36364 kg (maximum)
Tank Mass	2361 kg
Initial T/W_M	1.76 (as a MCSV)
Mass Fraction (APM only)	0.920
Total Mass (without payload)	39,518 kg
Total Mass (with payload)	50,000 kg

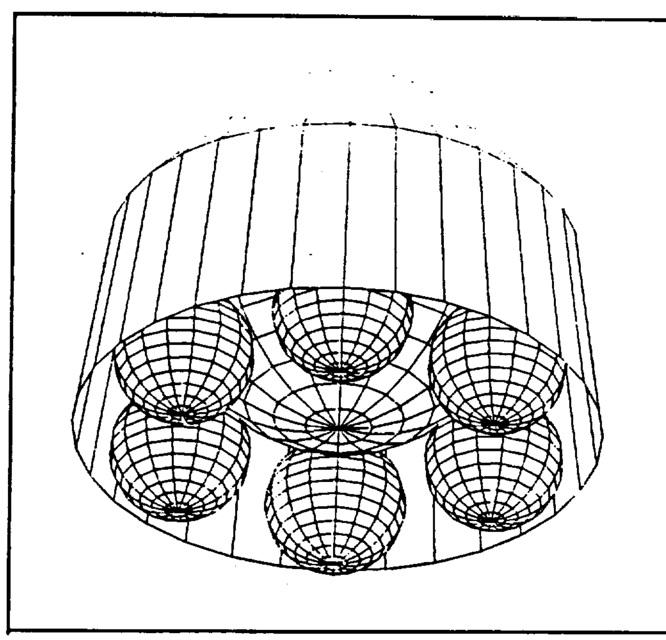
Figure 3.2.1.5.3.1-1 Ascent Propellant Module

MPV. On the trip from Earth to Mars the ECCV is not assembled because its crew cab is one and the same as the crew cab being used on a Mars excursion vehicle. The Earth Entry Aerobrake (EEA) is carried on a hinged fixture on the top structural ring of the MPV. After the excursion vehicle returns to the MPV (say from Deimos) the AOTPM is jettisoned leaving just the CCM. The rigid brake then automatically rotates into position and attaches to the base of the CCM. No fluid or electrical connections are necessary. For nominal Earth returns the ECCV is never detached and is carried into LEO by the MPV's main aerobrake.

3.3 SYSTEM DESCRIPTIONS

3.3.1 Habitats

3.3.1.1 MPV Habitation Module—The MPV carries two identical space-station derived habitation modules that provide complete redundancy in case of a serious problem with one module. Figure 3.3.1.1-1 shows the internal layout of one of the MPV habitation modules. The axial orientation of artificial gravity causes the interior of each habitation module to be divided into five stacked circular rooms connected by an enclosed ladder passage



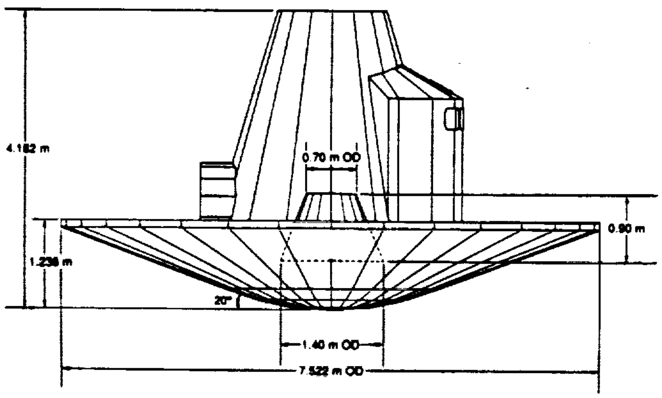


Figure 3.2.1.5.5-1 Earth Crew Capture Vehicle with Dimensions

way with doors on each level. Defining the furthest deck from the vehicles center as deck one and the closest as deck five, the sensed gravity ranges from 0.29 of Earth's gravity on deck five to 0.46 on deck one. To maximize crew activity in the greatest gravity, the sleeping quarters, radiation storm shelter, and supplies storage are all located on decks four and five. The mid decks are for vehicle command and control, science research, and data processing. Deck one is for recreation, meal preparation, and exercise. This layout matches increasing crew activity with increasing gravity. Another advantage of the axial orientation is the reduced volume needed for passage ways. This is because a person climbing a ladder needs roughly one-third the cross-section area of a person walking in a hallway. Deck-to-deck emergency passage is also available by removing large central lift-out sections of the floors. The life support systems provide 90% closure for both water and oxygen cycles and limited internal plant growth experiments provide intermittent fresh fruits and vegetables.

Radiation protection is provided in one/half of each module on deck five with the other half set up as a stateroom. Cosmic and solar radiation protection is provided by walls lined with consumable stores which, after being consumed are replaced with waste material. The central docking hub, enclosed by liquid oxygen, hydrogen, and an attached excursion vehicle can also serve as a backup storm shelter.

3.3.1.2 Nuclear Thermal Rocket—Piloted Vehicle—The habitation volume in the NTR-PV consists of two 2/3 length space station modules connected at each end by universal docking modules to make an enclosed racetrack arrangement. Because the NTR-PV traverses to and from Mars in half the time of the MPV, it doesn't need artificial gravity. The interiors are similar to the space station habitation module with foot restraints, vertical sleeping orientation, and conventional "walking" passage ways. The environmental closure of the NTR-PV is 90% in both water and oxygen with no en-route food generation.

3.3.1.3 Crew Cab Crew Accommodation—The crew can gain access through two hatches, one located at the top and used with the MPV, the other located on the side and used for egress to Mars' surface in later missions and for emergencies. The flight deck has two orientations for the commander and pilot. For space operations they use zero-g restraints (small straps and footrest pads) and face the nose of the module looking out the two forward windows. For descent and ascent to Mars' surface they sit in high-g seats oriented to view out the side windows.

3.3.2 Propulsion Systems

3.3.2.1 MPV and MCV Propulsion—The MPV or MCV has two propulsion systems: A storable RCS and a cryogenic main propulsion system. The RCS has an Isp of 310 seconds and uses monomethalhydrazine and nitrogen tetroxide as fuel and oxidizer. The tanks are located on the ends of each habitation module near the largest clusters of thrusters and hold a total of 12.82 tonnes of propellent. These motors orient the vehicle, spin it up for artificial gravity, despin it for aerobraking, precess its angular momentum, and control the spin rate during aerocapture. All thrusters are Marquardt R-4Ds with 444 N thrust except the aerocapture thrusters, which have 20,000 Newtons thrust each.

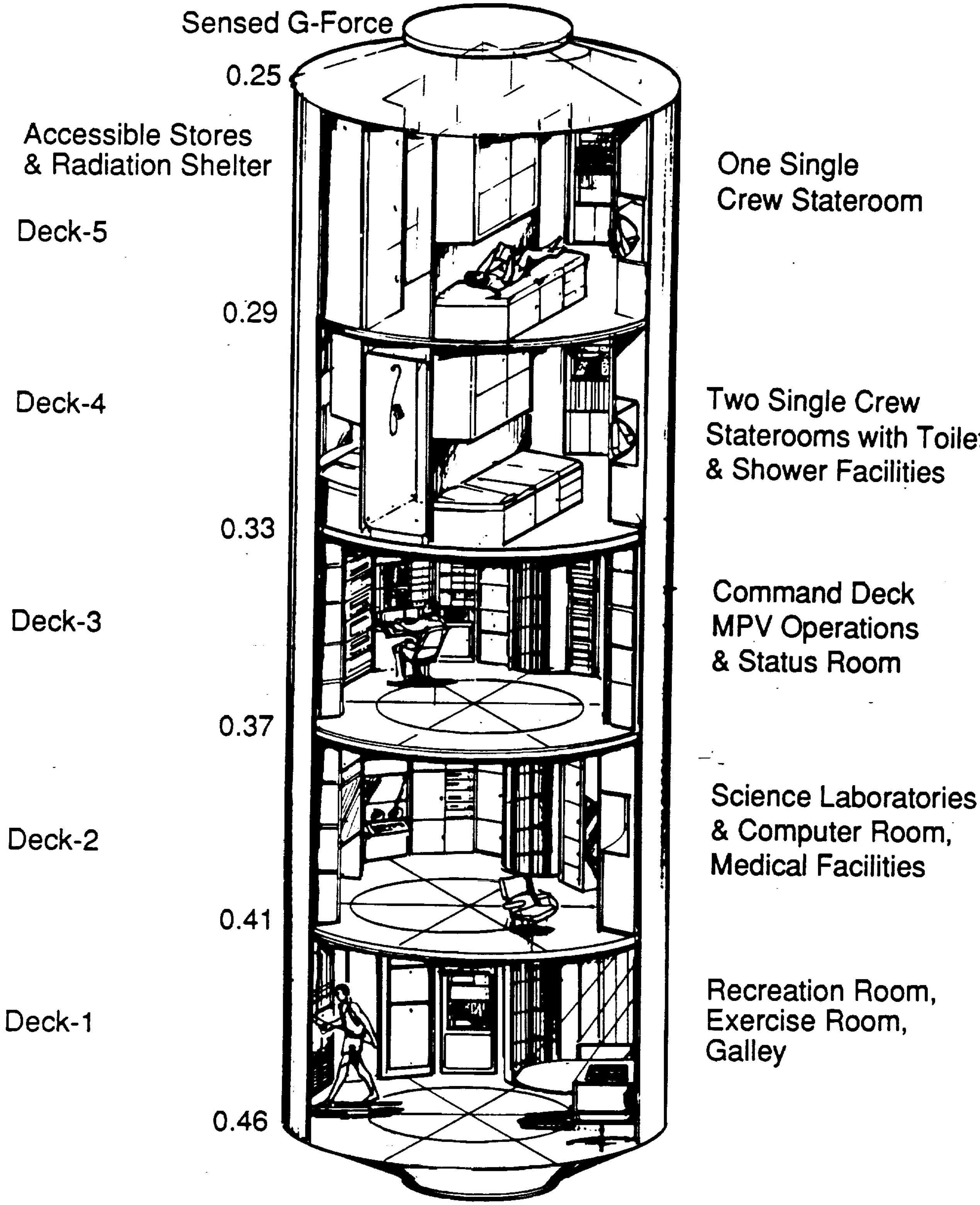


Figure 3.3.1.1-1 Internal Layout for Artificial-G Habitation Module

The cryogenic main propulsion system consists of liquid hydrogen and liquid oxygen stored in tanks that use multilayer insulation (MLI) and passive heat flow from the oxygen to the hydrogen. Because no reliquefaction is employed, boiled-off hydrogen ranges from 3 to 10 tonnes. Oxygen temperature is maintained by controlled conductivity with the hydrogen fuel. Propellant from these tanks feeds three RS-44 class engines (10.6 MPa, expander cycle) that have 625:1 expansion ratios giving them an Isp of 480 seconds. They have relatively low thrust at 33,400 Newtons each but they are sufficient for mid-course corrections, orbit circularizing at Mars, and TEI. The three engines are mounted on a modular truss structure that employs a single feed/pressurization/control interface similar to the space shuttle's interface with the external tank.

3.3.2.2 NTR-PV Propulsion—The NTR-PV has two 140 tonne loaded hydrogen tanks that use multilayer insulation and have tapered end domes facing the reactor to maximize propellant shielding of the crew as the propellant is drained. The hydrogen is fed into a 3000 megawatt reactor that generates 668.3 kN of thrust. This system has an Isp of 900 seconds at a chamber pressure of 4.3 MPa. Radiation to the crew is less than natural sources as a result of the combined effects of the short burn times (low inventory of radioactive elements), 60 meter reactor-to-crew separation, a 10 tonne disk shield, and the two shielding tanks of hydrogen. Table 3.3.2.2-1 lists the component masses of the NTR-PV propulsion system.

Table 3.3.2.2-1 NTR-PV Mass Breakdown

Reactor	10.0
Shield	10.0
Propellant	225.0
Tankage	25.0
Aeroshell	25.8
Payload	60.0
Total	355.8 Tonne

3.3.2.3 NEP-CV Propulsion System—The NEP-CV has four spherical tanks that contain 167.6 tonnes of argon. Multilayer insulation and reliquefaction pumps prevent any loss caused by boiling. The argon is fed to a four-by-four array of 500 kW ion thrusters operating at an Isp of 6000 seconds which produce a total thrust of 125 Newtons. Thruster lifetime is 10,000 hours, which means most of the 16 thrusters will be burned out by mission's end.

3.3.2.4 AOTPM Propulsion System—The AOTPM stores liquid hydrogen in two spherical tanks and liquid oxygen in two smaller spherical tanks. These tanks use multilayer insulation and oxygen-to-hydrogen heat flow to minimize hydrogen boil-off and eliminate oxygen boil-off. Thrust is provided by two RL-10B-2 engines. These engines have an Isp of 460 seconds and produce a total thrust of 196 kN. The outer shell of the AOTPM also provides the micrometeoroid shield necessary when the AOTPM is based at Phobos as the Phobos upper stage.

3.3.2.5 LAPM Propulsion System—The LAPM stores hydrogen in six spherical tanks, each with 1/3 meter of MLI. This gives an outside diameter of 2.7 meters. Oxygen is stored in a central sphere 3.0 meters in diameter including 1/6th meter MLI. These tanks feed six RS- 44 class engines with smaller, 225:1 expansion ratio engines. This ratio gives an Isp of 463 seconds and is smaller than the MPV/MCV engines to enable smaller doors in the LAPM's aeroshield and to reduce pressure losses when they fly in Mars' atmosphere. These engines produce 66.7 kN thrust each allowing a maximum Mars thrust-to-weight ratio of 1.7 just before touchdown.

3.3.2.6 APM Propulsion System—The APM has no engines. It carries propellant to allow the LAPM to ascent back to LMO. Hydrogen is stored in a single 5.88-meter diameter tank with 1/3 meter MLI on it. This tank absorbs heat from the six oxygen tanks at its base so they don't boil. The hydrogen heat is rejected with reliquefaction heat pumps that run off a surface-based power source. In addition to the MLI, both hydrogen and oxygen tanks also use vacuum jacket barriers to reduce the landed heat flow from Mars' atmosphere. The tanks are fitted with autogenous gas, feed, and fill lines. All oxygen tanks are manifold into a single feed line. These lines are at its base and connect directly to the LAPM when the vehicle is manufactured.

3.3.2.7 Crew Cab Module Propulsion System—The CCM carries 805 kg of monomethalhydrazine and nitrogen tetroxide as fuel and oxidizer for a propulsion system designed to provide a total of 500 m/s Δ-V. Thirty Marquardt R-4D thrusters give the CCM double fault tolerant reaction control authority and the ability for limited trajectory correction or orbit raising maneuvers. The tanks, regulators and feed/fill lines are located on the service deck, below the crew carrier deck.

3.3.2.8 Trans-Mars Injection Stage (TMIS) Propulsion System—The TMIS is common with the third stage of a 10 meter cargo diameter Shuttle-Z heavy lift launch vehicle. It has a single hydrogen tank that holds 18,140 kg of hydrogen and is configured with a barrel section and two domes. Below the aft dome is a ring of eight barrel/dome oxygen tanks that contain 108,860 kg of LOX. Inside of this ring is a single derivative Space Shuttle Main Engine (SSME) that has an Isp of 471 seconds and a nozzle expansion of 300:1 with the nozzle in the extended position. This engine generates 2372 kN of thrust.

3.3.2.9 Phobos Tether/Propulsion System—Although a tether does not appear to be a propulsion system in itself, it's application does reduce the requirements for traditional rocket propulsion systems and, therefore, it is classified as such. By placing a 1400 km tether on Mars' moon Phobos, a vehicle can begin at Phobos and be reeled out either toward Mars or away from Mars. Figure 3.3.2.9-1 shows the resulting orbits after tether release for both directions. The 1400-km length was selected because that is the length required for a 30 km release periapsis in Mars atmosphere where aerobraking can be applied to continue removing orbital energy. This allows either an MCSV or MCL to go from Phobos to the surface of Mars with a ΔV of less than one kilometer per second. This same tether is also used in the opposite direction to add energy to an Earth-bound vehicle without consuming propellant. The MPV can reach a 7400 x 30500-km orbit, which is only 400 meters per seconds shy of escape velocity.

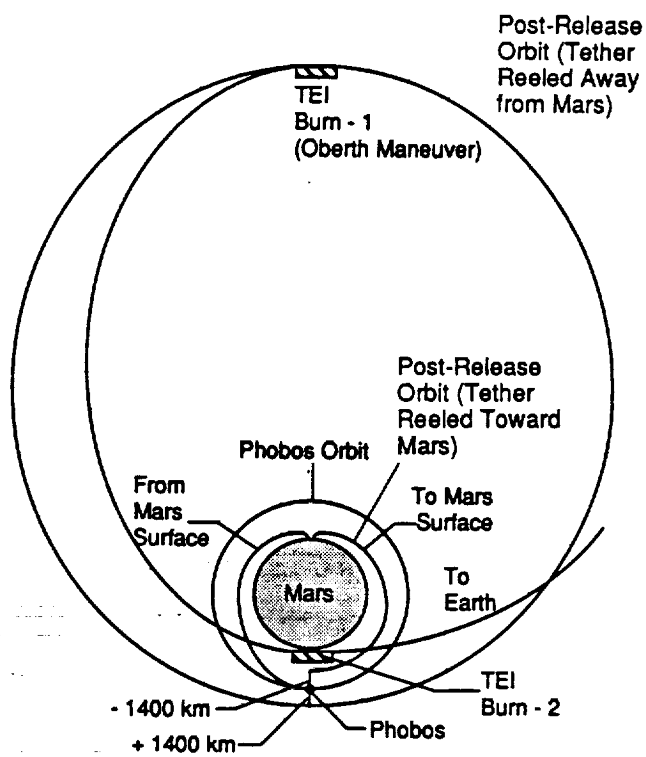


Figure 3.3.2.9-1 Orbital Mechanics Using a 1400 km Tether at Phobos

By only using the tether to reel-out vehicles away from Phobos, net power can be generated with every operation. In fact, about 500 kg of Phobos water can be processed into liquid hydrogen and oxygen with each use. Also, it takes 500 kW of electrical power to reel-in an MCSV in one day. Further, to reel-in a vehicle means it had to rendezvous with the tether's end, which is a risky maneuver because the end of the tether is accelerating relative to the spacecraft. All of these factors were combined to conclude that the best use of the tether is to reel-out a vehicle and release it and then winch-in the unloaded line in two days time.

For a single Phobos tether facility to operate up and down, it must be located on the "side" of Phobos (near Phobos' horizon as seen from Mars). From this location the tether will be drawn out parallel to the local horizon. To prevent the cord from touching Phobos' surface and to provide an elevated landing/docking platform, a structural tower is built as shown in Figure 3.3.2.9-2. This tower has two docking receptacles that are mounted on electric linear motors that provide the initial 20 m/s velocity needed to swiftly escape Phobos and begin using Mars' gravity gradient to continue accelerating away. Two tethers are shipped to Mars for redundancy with only one operating at a time. Table 3.3.2.9-1 gives an accounting of the tether system components and masses.

3.3.3 Aeroassist Systems

Four different aeroassist systems are employed in the Mars Evolution case study. The MPV/MCV use a 39 meter 0.2 L/D umbrella type brake. The LAPM

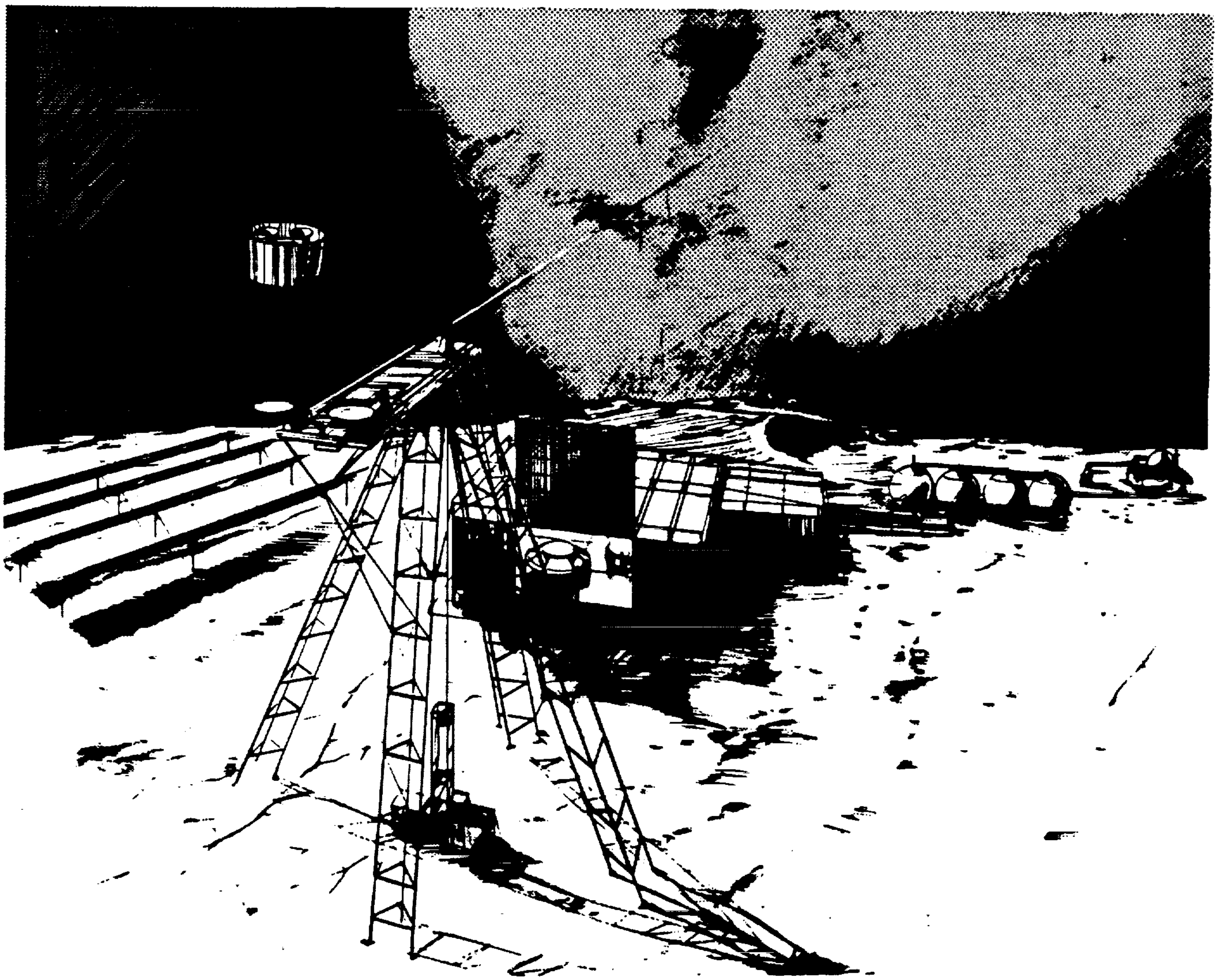


Figure 3.3.2.9-2 Phobos Tether and Propellant Facilities

uses a similar brake 23 meters in diameter. The NTR-PV uses a high aspect ratio raked ellipse brake that is assembled from three pieces in orbit. The last brake is identical in shape to the MPV brake but is only 7.5 meters in diameter and is made of one solid structure.

Table 3.3.2.9-1 ***Breakdown of Tether System Masses (Metric Tonnes)***

Total Tether System Mass: Two Tether Cords (1400km, 77t Capability) 30.914t Each: Based on Cortland Data on 20,000N Kevlar Lines Including a 2.0 Safety Factor		61.80	75.0
Dual Capstan Reel Storage (x2)		1.50	
Control System (Total Weight) Sensors, Computers, Controllers Tension Capstans and Structure	 0.75 1.00	1.75	
Dual Motors and Transmissions (10kW)		2.00	
Power System (10kWe PVPA)		0.70	
Micro Meteor Shield		0.75	
System Structure		1.00	
Landing/Departure Tower Beams, Rocket Anchors, Power Lines Accelerator Ramps, Beacons, Landing Pads		5.00	
Track to Vehicle Changeout Building		0.50	

The MPV/MCV employ a 39 meter diameter flex-fabric design. It consists of a 9.9-meter diameter central rigid core with 50 radial graphite polymide I-beams that fold out like an umbrella opens. This number of beams is driven by the tensile strength of the flexible ceramic cloth, known as Tailorable Advanced Blanket Insulation (TABI), which is derived from the space shuttle's Flexible Resuable Ceramic Insulation (FRCI). Elbow struts lock the I-beams into place and provide compression load paths that greatly reduces the I- beam bending moments, and hence, mass. The aerobrake and its support structure weigh 26 tonnes.

The MPV/MCV aerobrake is designed to the worst case entry conditions occurring during the 2004 opposition Mars encounter, where the entry C_3 is 60 km^2/s^2 which translates to 9130 m/s relative to the atmosphere. The brake's minimum diameter is determined by the maximum heating rates tolerable to the TABI. The structural mass is determined from the maximum acceleration and total braked mass. The lift-to-drag for blunt brake shapes is determined by the angle-of-attack, which is limited by wake impingement considerations behind the brake and by the need to keep the stagnation point on the solid core of the brake that uses FRCI tiles that can handle the higher temperatures.

Figure 3.3.3-1 shows the acceleration profile during a Mars entry for a C_3 of 60 km^2/s^2 and a lift-to-drag ratio of 0.2. The vehicle is flying along the top of its corridor, lifting downward to prevent skip-out. Note that the maximum acceleration is 4.3 Earth-g. Compare this to the 8.7 Earth-g of Figure 3.3.3-2 when the vehicle is flown along the bottom of the corridor in a lift-up orientation. Figure 3.3.3-3 shows the heating rates as a function of ballistic coefficient, which for the MPV is 124 kg/m^2. These data apply only to the first pass of the two-pass aerocapture. Initially, the vehicle captures into a 4-day loose ellipse after which the second aeropass lowers the apoapsis to that of Space Station Freedom's.

For the 2005 conjunction mission the nominal return trajectory uses a multirevolution return, which has an entry C_3 at Earth of 60 km^2/s^2. All other approach energies are less than 25 km^2/s^2. An aerobrake designed for the C_3=60 return would weigh 52 tonnes and would greatly increase the amount of propellant needed for all missions; hence, for an early return on the 2005 mission the MPV will be lost and the crew will return directly to Earth's surface in the ECCV. This will require building and launching a replacement MPV for the 2011 mission. Waiting 300 more days alieviates this problem because the C_3 is reduced.

The MPV/MCV aerobrake is launched folded up into a 10-meter diameter by 15-meter length package as seen in Figure 3.3.3-4. Upon orbit insertion and placement on the assembly fixture, the brake is unfolded to its flight configuration as shown in Figure 3.3.3-5. The brake is used at both Mars and Earth, making these vehicles reusable. On return to Earth the brake must be inspected and repaired

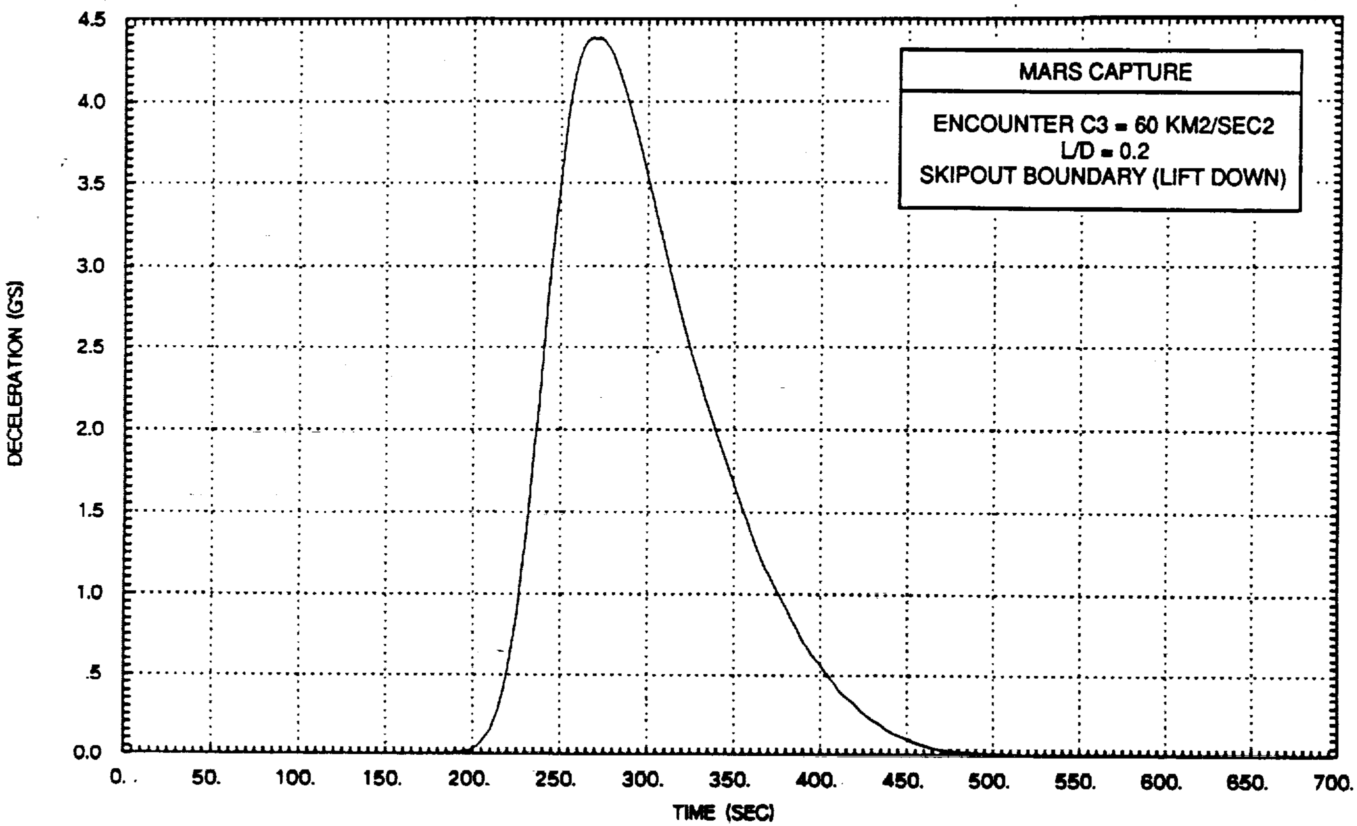


Figure 3.3.3-1 Mars Aerocapture Deceleration Profile Along the Top of the Corridor

before the next mission. This will consist of a telerobotic television inspection for burn-throughs, embrittlement, opened seams, deformed beams and struts, and fabric debonding from the structure.

The LAPM shown in Figure 3.2.1.5.2.1-1 uses a deployable brake design like that of the MPV/MCV. Its core section is 9.91 meters in diameter with 7 meter struts. It uses the same materials and construction as the larger brake but is also capable of folding back to the stowed position after use. When a LAPM is used in the MCSV the aerobrake is used repeatedly and is serviced at the gateway. Figure 3.3.3-6 shows the descent acceleration profile of a MCL with a 50-tonne payload. Note that the acceleration never exceeds twice that of Earth's. During ascent of the MCSV the brake folds up against the side of the APM and fits in under a built-in lip that prevents the oncoming atmosphere from forcing the brake open.

The NTR-PV uses a raked ellipse brake that is 81-meters long by 12.5-meters wide. It consists of an advanced composite structure coated with advanced ceramic thermal blankets. It is launched in three lengthwise pieces and integrated at the assembly fixture in LEO. The high length-to-width ratio provides greater lift-to-drag and, hence, lowered G-forces than a disk-shaped brake. This is important because a fast transfers mean high entry speeds and G-forces if downward lift is not available to prolong the deceleration.

The ECCV uses the same shaped brake as the MCV/MPV but does not need deployed beams and ceramic fabric because of its smaller diameter. It has three attachment fittings that affix it to the CCM. No electrical or fluid connections are necessary. Two parachute pods, one on each side of the CCM, deploy parachutes for the terminal descent to an Earth splashdown. These pods use spring loaded chute ejectors that fire at a 20 degree angle outward, ensuring that the parachute intercepts the atmos-

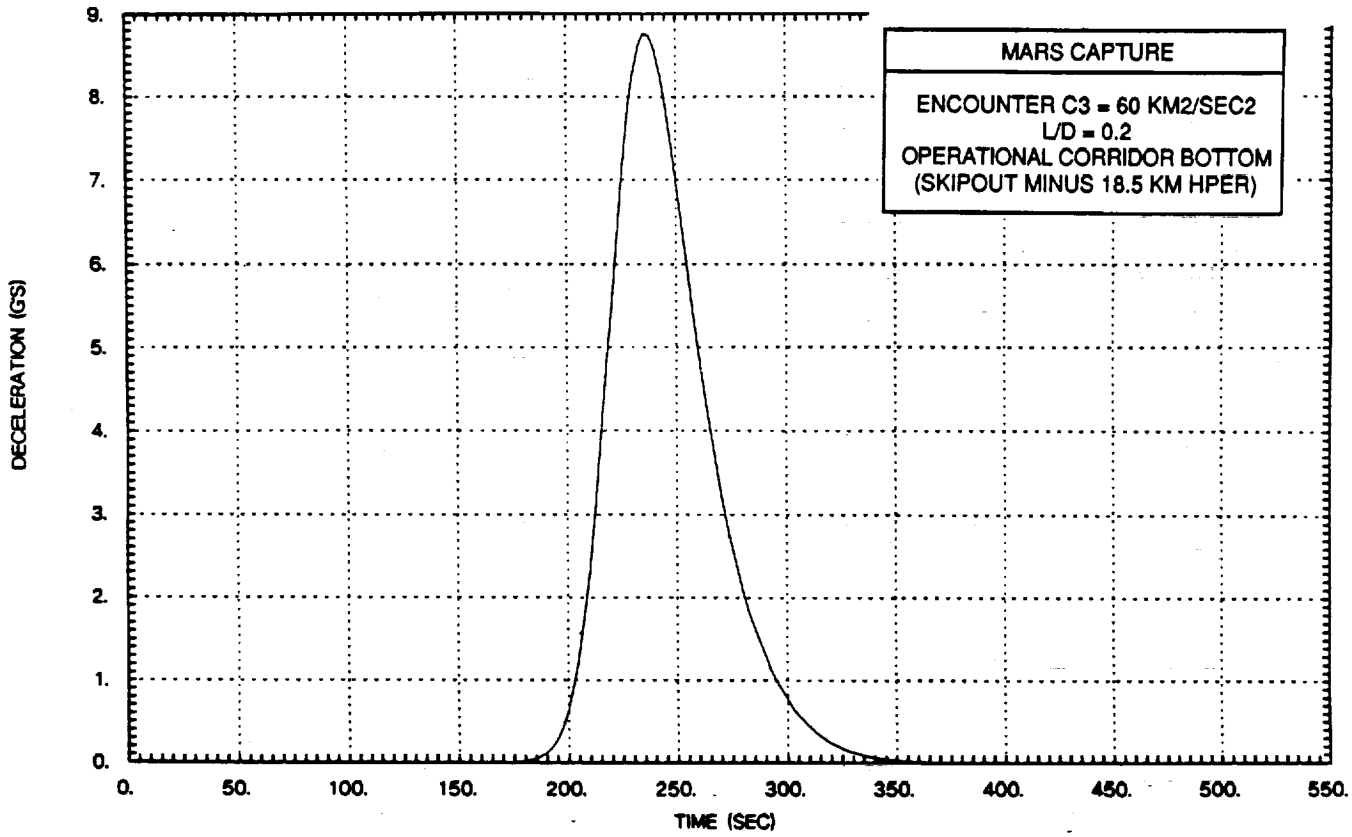


Figure 3.3.3-2 Mars Aerocapture Deceleration Profile Along the Bottom of the Corridor

pheric free stream. Upon landing, gas charges fill several buoyancy bags about the perimeter of the CCM.

3.3.4 Communications Systems

The communication system is capable of transmitting at least 5 Mbps from any vehicle to Earth; either directly or through one of two ariansynchronous relay satellite. For 10 percent of the time 15 Mbps can be transmitted and by combining four 34 meter receiving antennas 40 Mbps can be transmitted to Earth for critical phases or emergencies. The uplink capability far exceeds the required 20 Mbps even at 2.5 AU distance. To achieve this capability eight antennas of various size are mounted on the MPV, MCV, NTR-PV, and NEP-CV. A 5-meter diameter dish is cantilevered from the central ring frames and has fine pointing servos that maintain 0.035 degree pointing. The MPV spins like a wheel to generate artificial gravity and therefore must be Earth pointed to provide an uninterrupted link. A one meter medium gain antenna provides backup to the 5 meter dish at lower data rates. Six hemispherical radiation pattern antennas, three in the plus-Z direc-

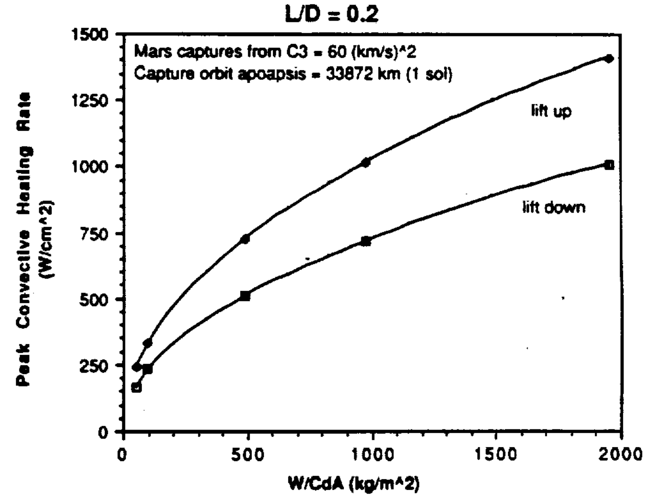


Figure 3.3.3-3 Convertive Heating Rates vs Ballistic Coefficient

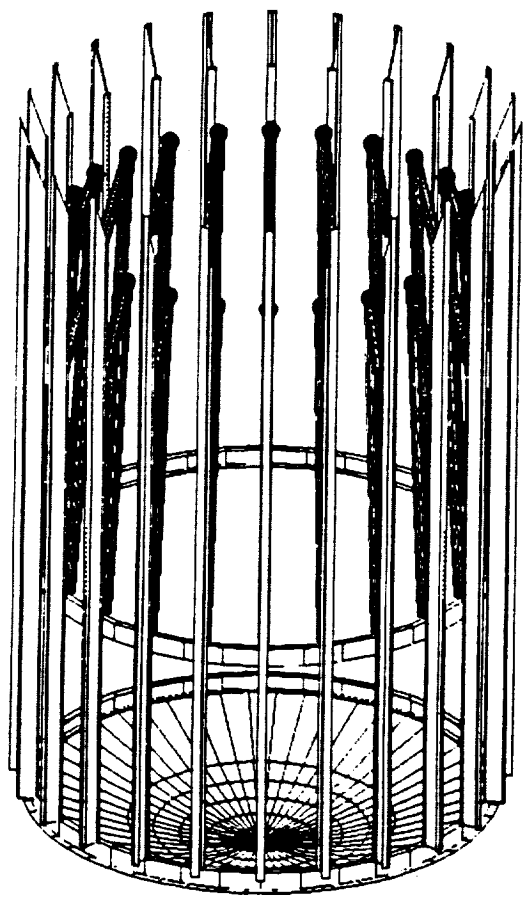


Figure 3.3.3-4 Flexible-Fabric Aerobrake–Launch Configuration

tion, three in the minus-Z, provide dual fault-tolerant communication, even if the vehicle's orientation is abnormal. The three minus-Z antennas are mounted on a swing arm that allows them to radiate and receive beyond the rim of the aerobrake antenna. Through these antennas 10 bps can be transmitted to Earth from 2.5 AU even if the vehicle is tumbling. Also, when the interplanetary trajectory is near Earth the low-gain antennas can be used to allow better orientation for solar array power generation and thermal control. Tables 3.3.4-1 and 3.3.4-2 lists the data requirements of the MPV.

At Mars two relay satellites are used to improve excursion vehicle and landed facility connectivity with Earth and support vehicle-to-vehicle communications.

3.3.5 Power systems

The Mars Piloted Vehicle employs solar panels and rechargeable batteries. Radial mounted panels that attach to the elbow struts of the aerobrake provide 20 kW at Mars when pointed directly at the sun. When Earth pointing requires the sun to be off-axis, panels mounted between the middle and upper ring frames make up for the lost power. Figure 3.3.5-1 is the angle between the sun and Earth during the 2004 and 2005 opportunities. The only times when the angle is near or exceeds 90 degrees is when the vehicle is near Earth or Venus. During these period the vehicle turns it pointing direction away from Earth so that power can be generated and low-gain antennas are used for communication. Reasonable data rates are maintained because of the close range between Earth and the vehicle.

The Mars Cargo Vehicle power system is the same as the MPV except for a reduction in the number of panels and batteries needed. The MCV can generate 5 kW at Mars and more when closer to Earth.

The NTR-PV generates its power with two deployable solar arrays that extend from containers located on the tops of the habitation modules. The panels have two-axis articulation for constant sun-pointing. Each of the two panels are 5 by 15 meters giving a total area of 150 square meters and a power of 10 kW at Mars. Rechargeable batteries provide power during aerocapture and when occulted.

On the Nuclear Electric Propulsion Cargo Vehicle, a power processing unit converts 100 kW of the high frequency alternating current coming from the Brayton cycle power conversion generator into 100 volt direct current power for the vehicle's use and for payloads being carried. Rechargeable batteries store 500 MJ giving a one day supply at 5000 W.

The CCM has a battery system that is rechargeable either by the MPV when docked, or by the excursion vehicle it is attached to. Storage capability is 86 MJ, which provides 1 kW for 24 hours. If and when the CCM is used in an emergency as the

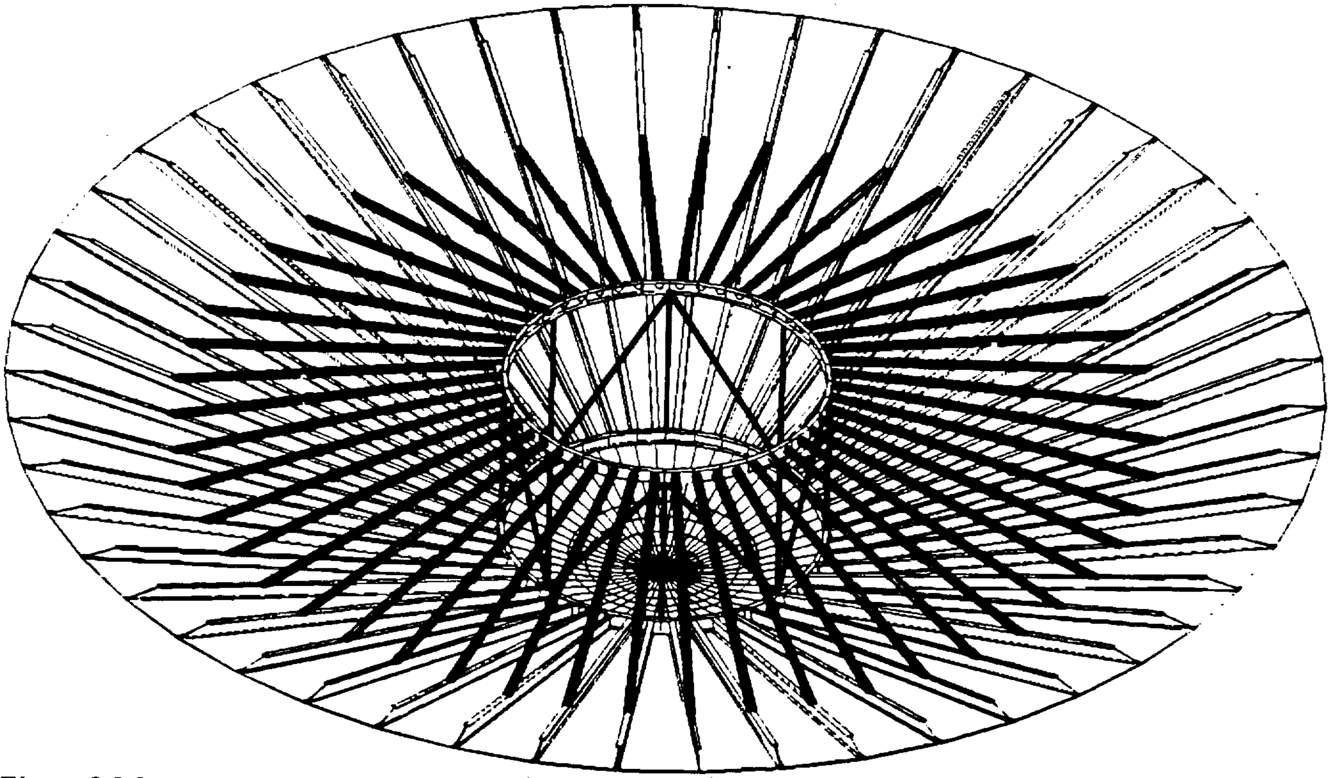


Figure 3.3.3-5 Flexible-Fabric Aerobrake–Deployed Configuration

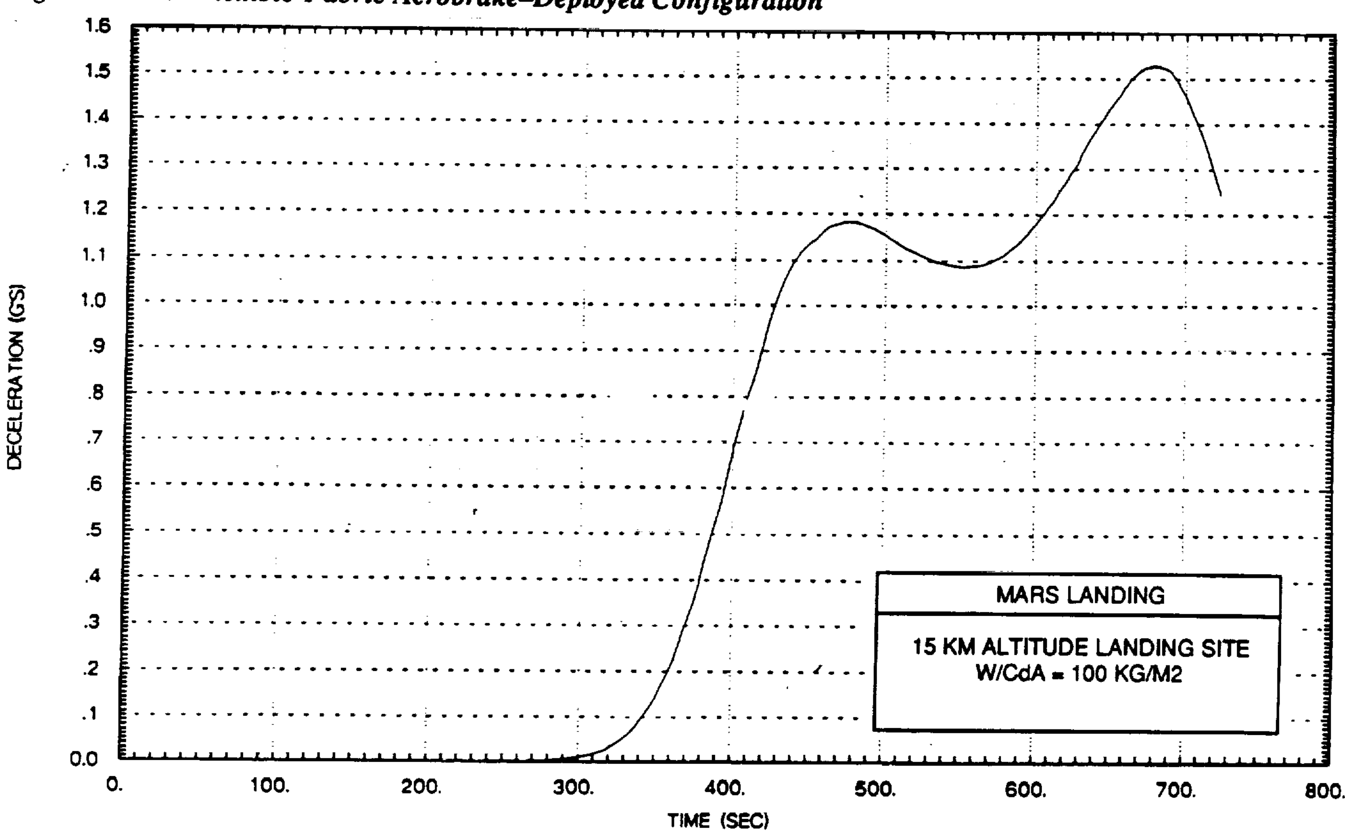


Figure 3.3.3-6 Mars Entry and Landing Acceleration Profile

Table 3.3.4-1 High Rate Data Requirements for the MPV

Video

Data Source			Duty Cycle								
	kbps/30	Max	Continuous			10%			1% & Spec/Emerg*		
Purpose/Type	HzFR**	BER	No. Ch	FR	kbps	No. Ch	FR	kbps	No. Ch	FR	kbps
Human Factors and C&C											
– Teleconferencing Quality	1,500	10^{-3}	2	30	3,000	3	30	4,500	5	10	2,500
– HD Color, Compressed, Med Rate	10,000	10^{-4}	–	–	–	1	3	1,000	5	3	5,000
Engineering Monitoring											
– B&W, Good Resol, Low Rate	3,000	10^{-3}	15	0.2	300	15	1	1,500	20	2	4,000
– B&W, Good Resol, High Rate	3,000	10^{-3}	2	1	200	2	5	1,000	5	30	15,000
– Standard Color TV Quality	1,500	10^{-3}	5	1	250	–	–	–	–	–	–
– HD Color, Low Rate	10,000	10^{-4}	–	–	–	–	–	–	12	1	4,000
– High Definition (HD) Color, Raw	100,000	10^{-4}	–	–	–	–	–	–	–	–	–
– Solar Monitor Video	3,000	10^{-3}	10	0.1	100	15	0.5	750	10	0.3	1,000
– Science (Imaging)	3,000	10^{-4}	8	0.5	400	16	1	1,200	–	–	–

Sound

Data Source			Duty Cycle								
	kbps	Max	Continuous			10%			1% & Spec/Emerg*		
Purpose/Type	per ch	BER	No. Ch	Ch Rate	kbps	No. Ch	Ch Rate	kbps	No. Ch	Ch Rate	kbps
Voice, Conversational Quality	20	10^{-2}	5	–	100	2	–	40	10	–	200
High Fidelity (Stereophonic CD Qual)	100	10^{-3}	1	–	100	3	–	300	5	–	500

Data

Data Source			Duty Cycle								
	Bits	Max	Continuous			10%			1% & Spec/Emerg*		
Purpose/Type	per ch	BER	No. Ch	Ch Rate	kbps	No. Ch	Ch Rate	kbps	No. Ch	Ch Rate	kbps
Engineering/Housekeeping Monitoring											
– Nominal Criticality		10^{-4}									
– Low Sampling Rate	12		250	1/s	3	250	10/s	30	250	100/s	300
– Medium Sampling Rate	10		100	10/s	10	100	100/s	100	70	1000/s	700
– High Sampling Rate	8		20	100/s	16	50	1000/s	160	25	10/s	2,000
– High Criticality	12	10^{-6}	50	10/s	6	100	100/s	120	200	100/s	240
Science Data											
– Stored/Buffered Data	2,000	10^{-4}	50	1/s	100	50	1/s	100	–	–	–
– Real-Time Data	2,000	10^{-4}	–	–	–	5	100/s	1,000	–	–	–
– Solar Flare/Radiation Monitoring	2,000	10^{-4}	30	1/s	60	30	10/s	600	30	100/s	600
Database Playback***	100,000	10^{-4}	1	1/s	100	10	1/s	1,000	30	1/s	3,000
Totals (Video+Sound+Data)			(544 Ch)		4.745 Mbps	(595 Ch)		13.4 Mbps	(530 Ch)		39.04 Mbps

* Spec/Emerg = Special and Emergency Use. See defivation of needs. (Note: Solar monitor reduced, unless rad emergency.) Assumes pointing, power, and communications systems healthy (see low-gain backup).
** Rate in kilobits per second for a nominal frame rate (FR) of 30 Hz. Bit stream is data compressed and encoded.
*** Checksum included

Note:

10% is 1.0 hr in the morning (at 7 a.m.), and 1.4 hr in the evening (6 p.m.)
1% is 7.2 minutes, twice per day (noon and nominally at midnight)

Table 3.3.4-2 Low Rate Data Requirements for the MPV

Video											
Data Source			Duty Cycle								
	kbps/30	Max	Omnidirectional			Broad-Beam			Burst*		
Purpose/Type	HzFR**	BER	No. Ch	FR	bps	No. Ch	FR	bps	No. Ch	FR	bps
Crew Status – Interior Cabin Views, Color Degraded	600	10^{-2}	–	–	–	–	–	–	2	0.1	4,000
Vehicle Status – B&W, Gross Resol, Low Rate (from External Monitors)	300	10^{-2}	–	–	–	–	–	–	5	0.02	1,000
Sound											
Data Source			Duty Cycle								
	kbps	Max	Omnidirectional			Broad-Beam			Burst*		
Purpose/Type	per ch	BER	No. Ch	Ch Rate	bps	No. Ch	Ch Rate	bps	No. Ch	Ch Rate	bps
Crew Status – Voice, Reduced Quality	3	10^{-2}	–	–	–	–	–	–	1	–	3,000
Vehicle Status – Minimum Quality Sound	0.5	10^{-2}	–	–	–	–	–	–	2	–	1,000
Data											
Data Source			Duty Cycle								
	Bits	Max	Omnidirectional			Broad-Beam			Burst*		
Purpose/Type	per ch	BER	No. Ch	Ch Rate	bps	No. Ch	Ch Rate	bps	No. Ch	Ch Rate	bps
Vehicle Status – Vital Monitors	2	10^{-4}	25	0.2/s	10	–	–	–	–	–	–
– Nominal Criticality		10^{-3}									
– Low Sampling Rate	12		–	–	–	125	0.01/s	15	250	0.1/s	300
– Medium Sampling Rate	10		–	–	–	20	0.1/s	20	70	1/s	700
– High Sampling Rate	8		–	–	–	12	0.5/s	48	25	10/s	2,000
– High Criticality	12	10^{-4}	–	–	–	10	0.1/s	12	200	1/s	2,400
Totals (Video+Sound+Data)			(25 Ch)		10 bps	167 Ch)		95 bps	(545 Ch)		14,400 bps

* Omnidirectional and Broad-Beam (approx 1/2 steradians) are separate and independent antenna and drive systems. Burst mode utilizes high gain system whenever attitude determination system is consistent with orientation toward DSN receivers or when receiver detects uplink communications signal.

** Rate in kilobits per second for a nominal frame rate (FR) of 30 Hz. Bit stream is data compressed and encoded.

ECCV, then power must be used judiciously because the time between MPV departure and Earth arrival could be as long as a week.

The LAPM uses combined hydrogen and oxygen from a gas generator to create pneumatic power to vector the engines, to drive the engine/leg doors, and to extend the landing legs.

Both the Ascent Propellant Module (APM) and the Ascent and Orbit Transfer Propulsion Module (AOTPM) generate up to 20 kW with oxygen/hydrogen fuel cells. Exportable power is fed to the CCM or to surface equipment. The water generated is stored and either transferred back to the MPV for crew consumption or used as needed by the surface crew on Mars.

3.3.6 Thermal Systems

Cryogenic hydrogen and oxygen at 30 and 160 degrees Kelvin respectively present difficult thermal control problems for vehicles that first must travel interplanetary space from 0.7 to 2.5 AU, then descend into Mars atmosphere, and finally ascend

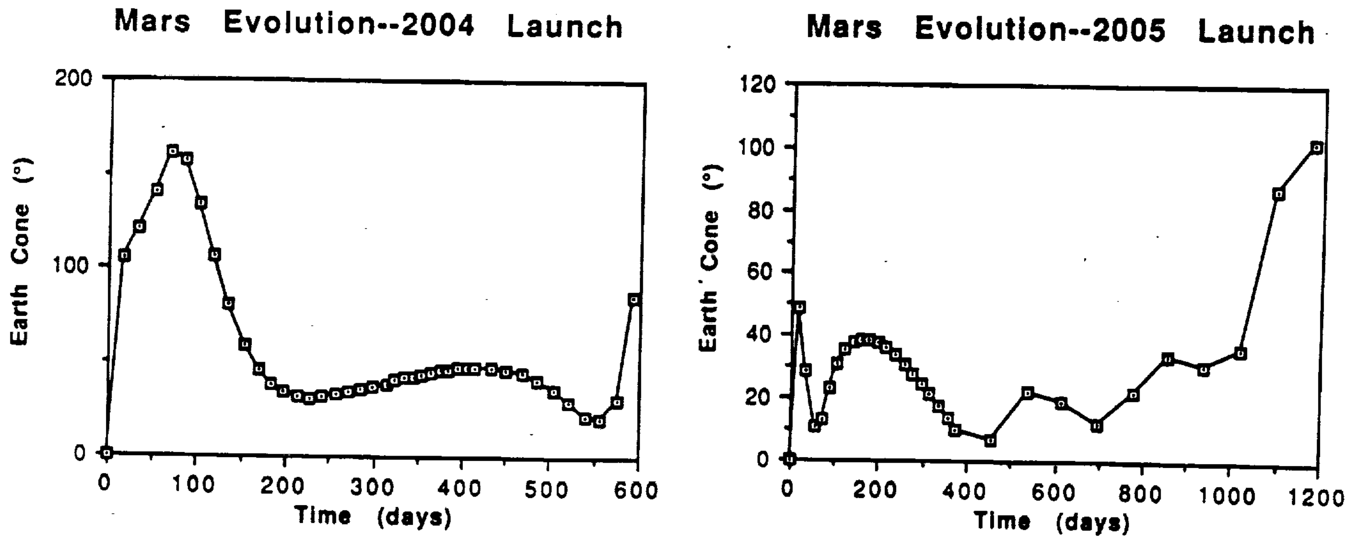


Figure 3.3.5-1 Sun-Spacecraft-Earth Angle for Missions One and Two

back into space after staying on the surface for up to a year. All vehicles have controllable conductivity between the hydrogen and oxygen tanks that provides a cold sink for the oxygen that offsets the heat input from solar, planetary and vehicle sources. The hydrogen tanks use 200 layers of MLI. The hydrogen tanks also incorporate vapor-cooled shields and, in the landing excursion vehicles, vacuum jackets to minimize convective heat losses into the atmosphere. Thermodynamic vents are employed to exchange heat in the retained hydrogen with the hydrogen being vented—minimizing the mass loss.

The people and electronic equipment in the habitation modules produce several kilowatts of heat that must be rejected into space. This is done with one of two radiator panels that are mounted orthogonally to each other on each habitation module. The spinning MPV and the varying sun-cone angles force the use of two radiators that are hinged along their long axis allowing them to be in-line with the sun where they absorb the least heat. During each revolution of the MPV the radiators must also go through a rotation cycle. When the solar-cone angle exceeds 45°the side mounted radiators on each habitation module are employed. These panels are also articulated but do not have to cycle as the MPV rotates.

All systems must be designed to receive a large surge of radiative heat from the MPV, MCV, or lander's aerobrake. For the habitation modules a low absorptance coating must be applied to the side facing the brake. For the propellant tanks the heat surge will result in a surge in boiled-off hyrdogen that is budgeted for at the outset.

3.4 OPERATIONS CONCEPT

3.4.1 ETO Manifest

For every kilogram of payload put into LEO, 30 to 100 kg of launch vehicle is required; hence, the Earth-to-Orbit scenario for the Mars Evolution Case Study is of great significance. Specific requirements have been defined to constrain the ETO lift capability to reasonable values. For any given year the total lifted mass should not exceed 570 tonnes and of that value not more than 90 tonnes can be hardware—although the hardware can be averaged over two consecutive years. Also, the ETO cargo vehicle has an assumed capability of 140 tonnes to a 500 km circular orbit and can only fly four times per year and not more frequently than every 45 days. Finally, a 5 man crew carrier vehicle can be flown twice per year.

Figure 3.4.1-1 is a breakout of each mission's mass by propellant, dry vehicle, and payload. Dry vehicle mass is hardware mass associated with the transportation system and payload mass is hardware or propellants that are being taken to either Phobos or Mars. Propellant mass includes cryogenic and storable bipropellants. As can be seen, flights one, two, four, and five have very low payload to propellant masses because they fly round-trip and leave very little at Mars. Mission-3 is the cargo mission and can replace TEI propellant with useful cargo giving it a much greater payload-to-propellant ratio. Mission-6 is the NTR-PV mission and uses three times as much propellant as the minimum required allowing it to traverse interplanetary space in half the time. Finally, in mission-7, the NEP-CV achieves a much greater useful payload ratio because of its 6000 second specific impulse thruster. This is at the cost of taking over two years to get to Mars.

Although several missions exceed 570 tonnes gross mass, they do not exceed the requirement of 570 tonnes per year because ETO flights span several years. Figure 3.4.1-2 shows both the launch mass per year and the number of 140 tonne cargo flights per year. The solid horizontal line defines the 570 tonne required limit. The missions were manifested by starting with the last mission and using up to four 140 tonne ETO flights per year if necessary. If more flights were needed then the previous year is used. It was assumed that the last ETO flight would carry any partial load. No packaging factors for geometry were considered and all flights were filled to the 140 tonne limit. This is a reasonable assumption if propellant scavenging is used. This scenario assumes that each flight rounds up it's payload to 140 tonnes by carrying oxygen in an oversized orbital injection stage tank. This oxygen is then transferred to a holding tank on the assembly fixture until needed before departure. Oxygen's -160° C boiling point, high density, and the fact that it is the great majority of total propellant mass make it a better choice than hydrogen for scavenging. On the last ETO flight a dedicated hydrogen tank is lifted to the assembly fixture and the mission departs soon after its arrival at the fixture. This is feasible because, except for the NTR-PV, all missions use less than 140 tonnes of hydrogen.

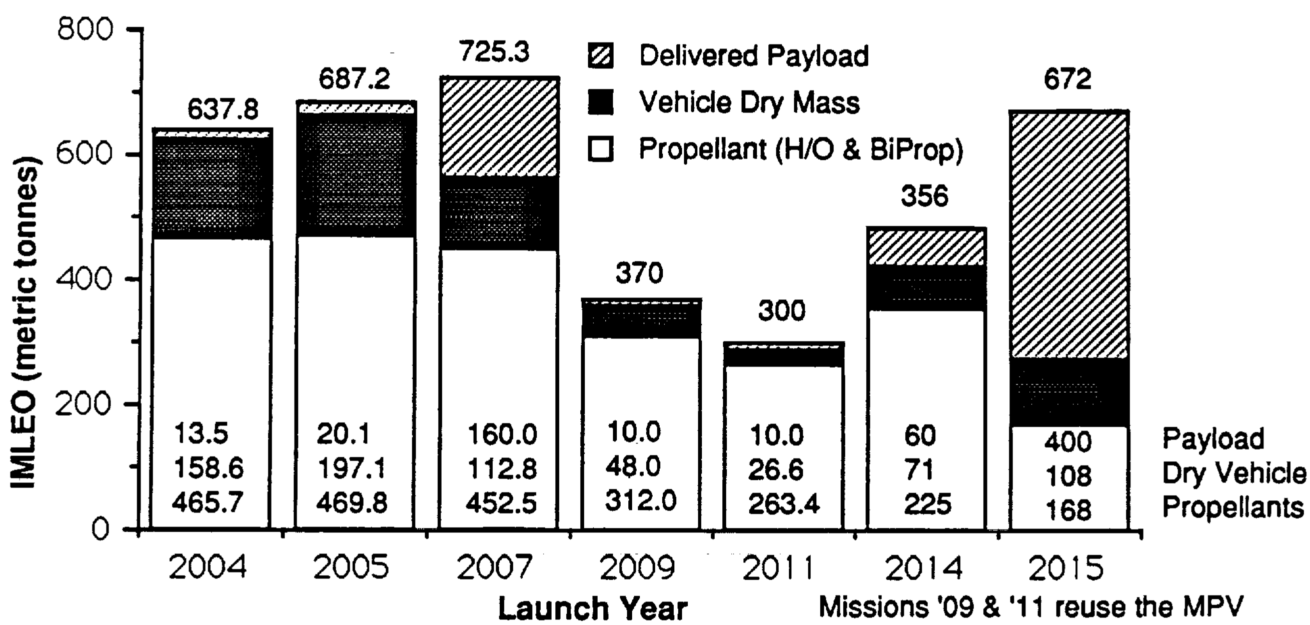


Figure 3.4.1-1 Breakout of Each Mission's Initial Mass in Low Earth Orbit

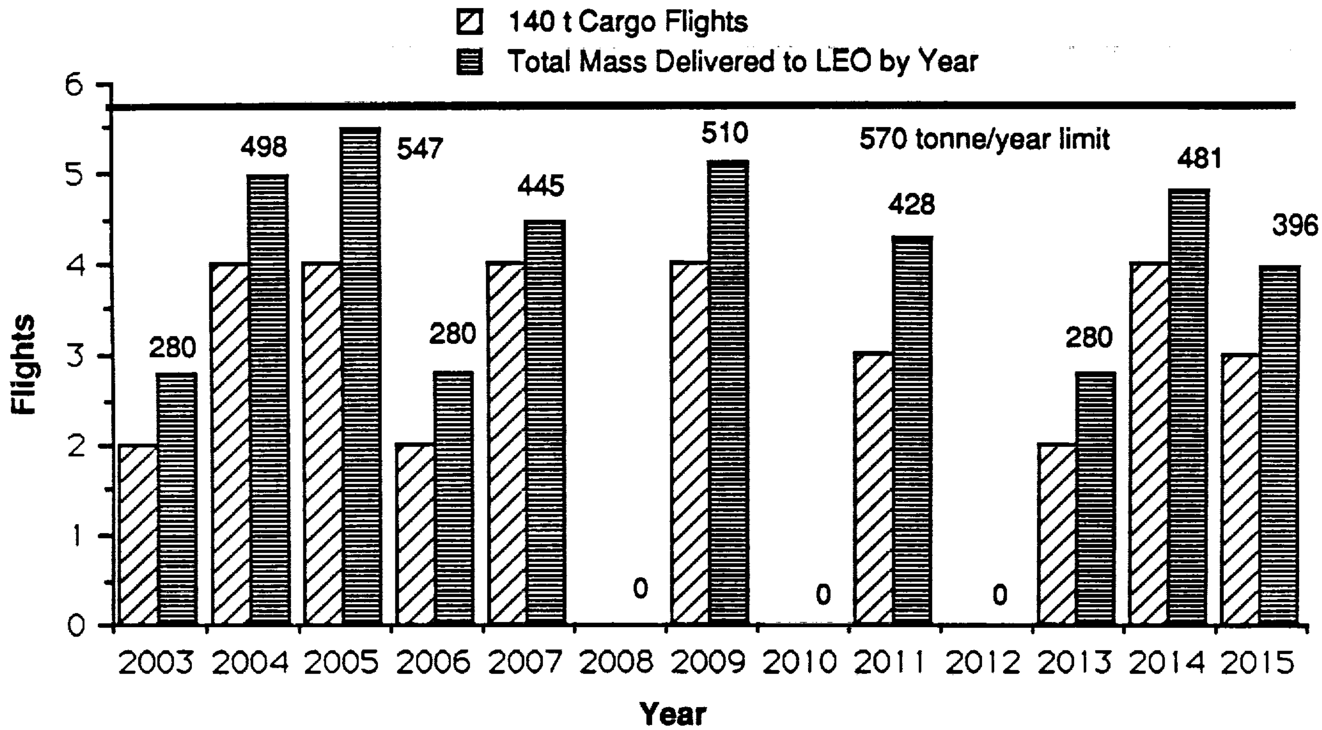


Figure 3.4.1-2 Launched Mass and Required Flights by Year for Mars Evolution

The heavy front end loading caused by missions-1 and 2 in 2004 and 2005 causes the 90 tonne per year hardware limit to be violated. To abide by that requirement pushes the first launch up to 2001, which was considered too early. By violating the hardware limit the first launch can be pushed back 2 years to 2003. Because the missions in 2009 and 2011 are roughly half the masses of the earlier missions, no previous year ETO flights are necessary which leaves 2008 and 2010 without ETO launches.

3.4.2 On-orbit Assembly

Figures 3.4.2-1 through 3.4.2-3 show the assembly sequence for the first mission. The assumed cargo launch vehicle is a 140 tonne capable Shuttle-Z and the piloted launch vehicle is assumed to be the Space Shuttle. The first Shuttle-Z flight lifts the MPV's aerobrake, integral TEI tankage, and central docking hub. Loaded into the TEI tankage is the flight oxygen and hydrogen that must survive high accelerations at Mars and provide long term storage; both of which can be tested during launch and storage before departure. As described before, excess oxygen is also carried to a depot tank at the fixture. On arrival, the Shuttle-Z third stage docks to the receiving docking berth on the assembly fixture and separates from the aerobrake. It is then carried by the fixture's manipulator arm to the TMI storage berths at the bottom of the truss where the excess oxygen is pumped into the holding tank. The stage is saved because it will become one of four TMI stages later on. The aerobrake is placed on the MPV berth and is unfolded, locked, and verified. The next cargo flight brings a host of smaller hardware elements such as the TEI engines, solar panels, communications antennas etc. The airborne support equipment (ASE) that holds these payload elements during launch is attached to the receiving dock. One at a time the components are removed from the ASE and remotely attached to the MPV ring frame and aerobrake structure. Again, excess oxygen is placed in the holding tank and the Shuttle-Z's third stage is berthed. The third cargo flight brings up the two habitation modules and more excess oxygen. After the habitation modules are attached a crew is launched on the Space Shuttle and EVA and IVA activities begin verifying the

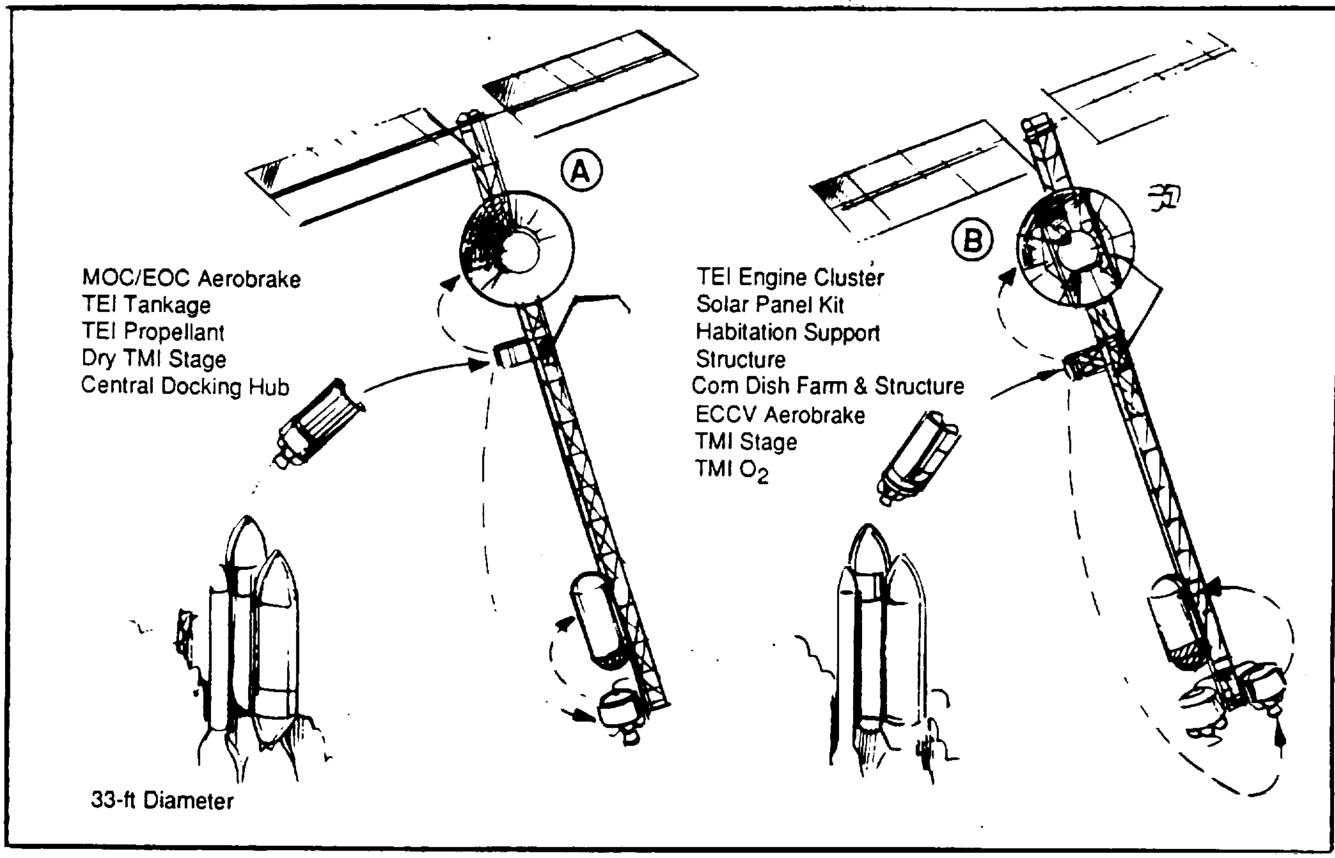


Figure 3.4.2-1 On-Orbit Assembly Sequence–ETO Flights One and Two

robotic assembly and performing more delicate mating activities directly such as propellant feed line attachment, antenna feed connections and solar panel electrical hook-ups. The fourth cargo flight lifts the Phobos/Deimos Excursion vehicle and the Mars Rover Sample Return (MRSR) payload as well as more excess oxygen. The Ph/DeEV is docked directly to the MPV and the MRSR is placed on a generic cargo platform attach point cantilevered off the middle ring frame on the MPV. The next two cargo flights lift the remaining oxygen. The final flight lifts all the hydrogen required for the TMI stages. The TEI hydrogen tank is also topped off with hydrogen.

After the MPV has been assembled it is moved by the arm to the integrating platforms where the TMI stages are attached without propellant. After all four TMI stages are attached, propellants are loaded through individual feed lines, drawing directly from the depot tanks. A Space Transfer Vehicle then attaches to the last stage and maneuvers the stack to a safe departure orbit where the high-expansion SSMEs can fire, initiating trans-Mars injection.

3.4.3 Mission Operations Sequences

All phases of the mission are primarily controlled by the Mission Commander in the MPV except for TMI, Mars surface operations, and after returning to Earth orbit. During TMI, control is with Mission Control on the Earth because of the complexity and risk of the multiple TMI stages and because the communications delay is trivial. Mars operations are led by the Mission Commander from the excursion vehicle (MDV or MCSV) going to and staying on the surface. In some missions the MPV is vacated in Mars orbit entirely. Upon returning to Earth orbit, the control then reverts to Mission Control on Earth. Table 3.4.3-1 shows the command authority and the backups during each mission phase.

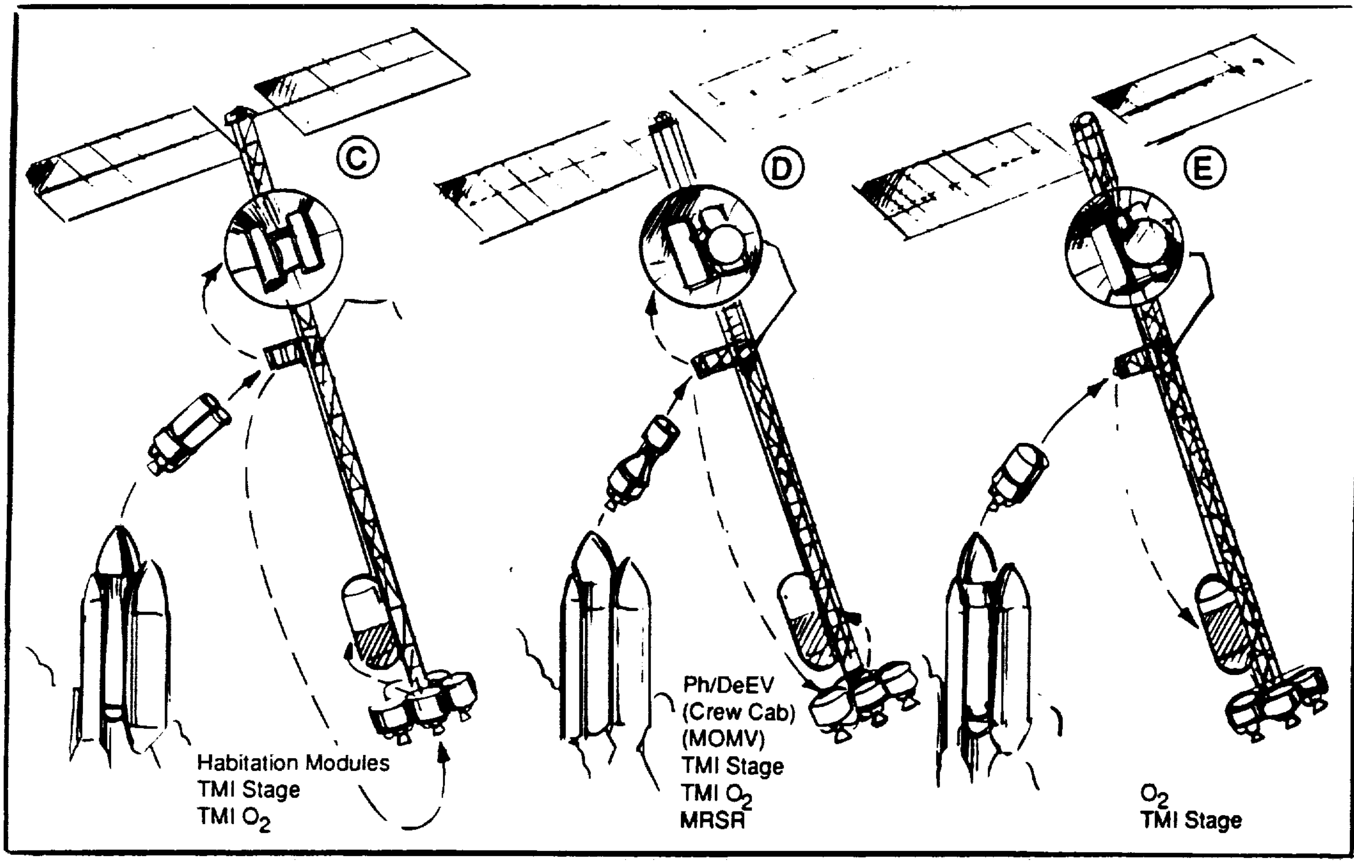


Figure 3.4.2-2 On-Orbit Assembly Sequence–ETO Flights Three, Four, and Five

Table 3.4.3-1 Mission Control Authority by Phase

Mission Phase	Control Primary	Secondary	Tertiary
On-Orbit Assembly	SSF	MC	None
Trans-Mars Injection	MC	MPV	None
Interplanetary Cruise	MPV	MC	None
Mars Aerobraking	MPV	MC	None
Mars Orbital Ops	MPV	MC	None
Mars Surface Ops	EVCD	MPV	MC
Trans-Earth Inject	MPV	MC	None
Interplanetray Cruise	MPV	MC	None
Earth Aerobraking	MPV	MC	None
Earth Orbital Ops	MC	MPV	SSF

Legend:

SSF	Space Station Freedom
MC	Mission Control (On Earth)
MPV	Mars Piloted Vehicle Control Deck
EVCD	Excursion Vehicle Command Deck

3.4.4 Reliability and Safety

Each piloted vehicle has several design features to enhance reliability and safety. The MPV has two identical habitation modules connected by pressurized tunnels during all mission phases. The crew has access to both modules, the central docking hub and any docked excursion vehicle. In the event a habitation module suffers an accident that renders it inhospitable and also blocks passage to the other module, the crew can exit through the hyperbaric airlock where they have direct control of the spin/despin thrusters. After despinning the vehicle they can EVA along hand-holds to either the excursion vehicle or the airlock on the other habitation module. Apart from the life-support safety features, the MPV has three TEI engines that are highly reliable and serviceable RS-44 derivative engines. These engines have preventative diagnosis telemetry out-

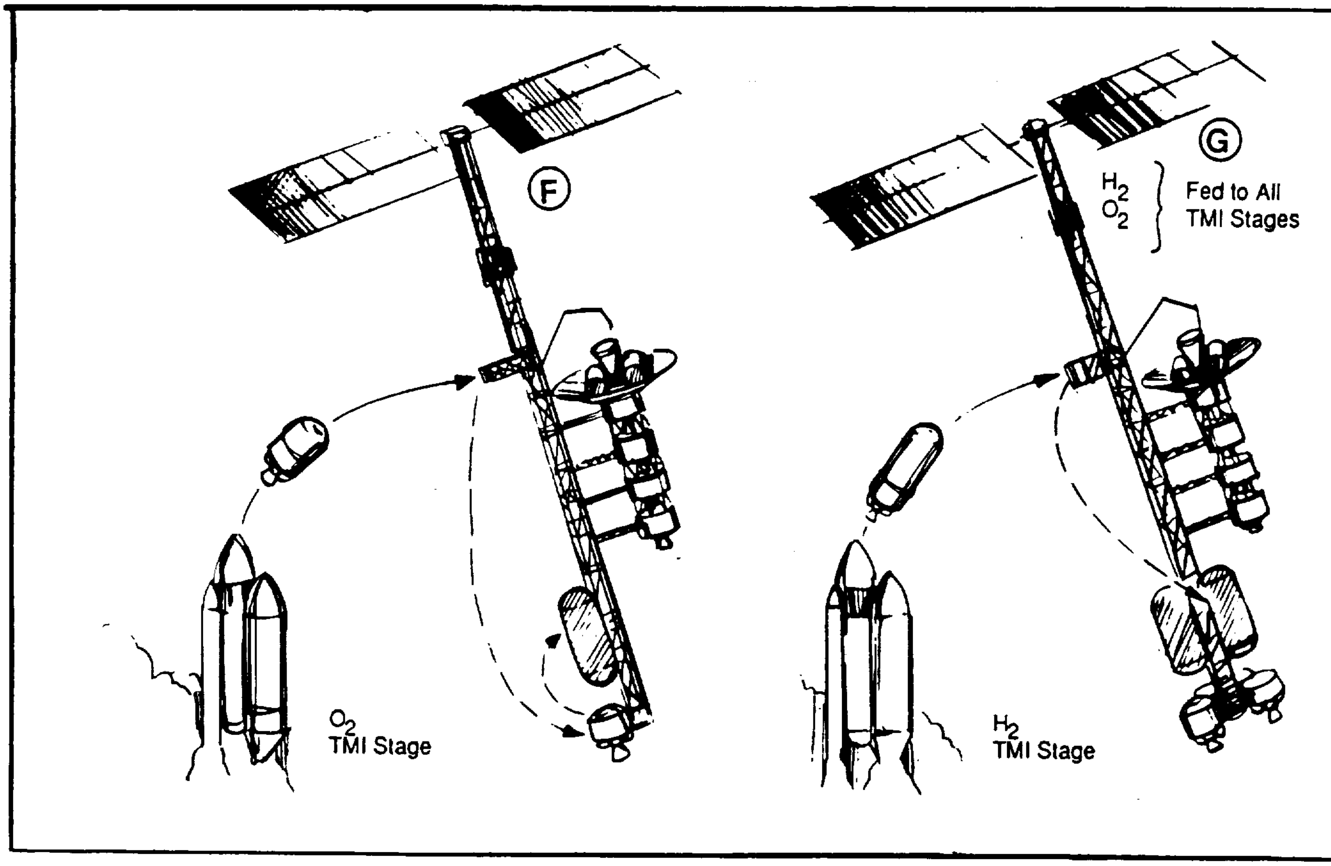


Figure 3.4.2-3 On-Orbit Assembly Sequence–ETO Flights Six and Seven

puts and are fired many times before committing the vehicle to capture into Mars orbit. Another safety feature is the Earth Crew Capture Vehicle. During Earth return, if a problem prevents the MPV from aerobraking, the crew can abort to the ECCV and, up to one week before entry, depart the MPV and enter directly into the atmosphere for an Apollo style splashdown in the ocean.

The Nuclear Thermal Rocket Cargo Vehicle (NTR-CV) safety features include twin habitation modules connected with a logistics tunnel. The rapid trip times of about 100-170 days is a significant safety factor in itself. Radiation protection from the reactor is provided by a 57-meter separation distance, a 10 tonne reactor shadow shield, and the hydrogen propellant remaining after TMI in the aft tank. The 3000 MWt reactor also operates for only a few minutes meaning very little radioactive inventory is built up and that inventory has more rapid decay times.

Safety and reliability are also prime concerns in the designs of the MDV and the MCSV. The MDV has the capability to abort its landing by staging away from the LAPM and ascending back into orbit. Additionally, the CCM can abort from the AOTPM if it is within 500 m/s of a stable orbit. The AOTPM has two highly reliable RL-10B-2 engines. The design cannot tolerate an engine failure at lift-off but it can achieve a stable orbit if the one engine fails after 80 seconds. To protect the hydrogen and oxygen tanks from micrometeor impacts all three modules of the MDV have outer fairings (for streamlining) that provide a displaced impact shield that effectively disperses the ejecta generated by an impact.

The MCSV, although a single-stage vehicle, also has an abort during descent capability with its six RS-44 derivative engines. During all propulsive phases of the descent the thrust-to-weight ratio is greater than 1.6. For both the MDV and the MCSV the abort could not occur until the aerobraking phase was completed. The MCSV does not have a stageable crew cab, so the entire vehicle must achieve a stable orbit. This reduced flexibility is offset by the LAPM and CCM's maturity gained from earlier mission experience and design improvement.

3.4.5 Useful Life

Each vehicle in the Mars Evolution Case Study has different useful life requirements depending on its planned use. The previous Figure 3.1.2-1 shows the disposition of all vehicles by following their location lines. MPV's one and two are used for two missions each, hence they need a 10-year useful life that includes time spent before launch and time in orbit before departure. After their first flights, the MPVs must be refurbished at the Assembly Fixture with a new ECCV, much of the habitation module's interior, any TEI thrusters not in perfect health, and any equipment damaged by micrometeors.

The MCV is not reused and therefore only requires a useful life of three years. This vehicle could be reused if the mission scenarios changed by adding more propellant for TEI at the cost of reduced payload capacity to Mars or Phobos.

The LAPM, AOTPM, CCM, EEA all start out being expendable and evolve into reusable modules. Only the APM begins as a reusable vehicle. In their expendable modes the useful life is 3 years for the LAPM and AOTPM. For the CCM and EEA, which must return to Earth, their useful lives are 5 years. For the later phases all of these modules become reusable and must have useful lives of at least 15 years to be practical as reusable vehicles.

3.5 DEVELOPMENT SCHEDULE

The Mars Evolution Case Study has a very ambitious development schedule driven by an initial mission in 2004 immediately followed by the overlapping mission in 2005. These two missions require the design, construction, launch, and checkout of two MPVs, two excursion vehicles, a new HLLV, and the ground control network necessary to manage these vehicles. The MPV program must begin in 1992 with a phase-A study to define the vehicle configuration followed by phase-B in 1995 to produce a detailed design. Following phase-B, construction begins in late 1997 and is completed in the middle of 2003. The number-2 MPV lags this schedule by one year through the build phases. Figure 3.5-1 details the development schedule for all the vehicles and excursion vehicle modules. In the following section the schedule of technology development is presented which is in concert with the need dates of the vehicle and module milestones.

3.6 TECHNOLOGY NEEDS

A principal objective for this case study is to maximize the use of advanced technologies that reduce the amount of propellant mass needed in Earth orbit. The most effective technologies are aerobraking, nuclear thermal rockets, nuclear electric propulsion and *in-situ* propellant production at Mars and Phobos. A 1400-km tether is also employed to transfer momentum between vehicles and Phobos. A tether of this length requires several unique technologies: the capability to produce a 1400 km continuous tether without flaws and a method of inspecting and repairing the tether once deployed. *In-situ* propellant production also requires supporting technologies. It requires development of low dust ore extraction methods from asteroid class bodies and the ability to remotely extract water, perform electrolysis on it, liquify the hydrogen and oxygen, and preserve these cryogenic propellants over time.

Some technologies do not reduce LEO propellant requirements but are still unique to this case study. The MCSV, which lands and takes-off repeatedly must have dust tolerant, hydrogen/oxygen engines with extensive health monitoring. Both of the nuclear interplanetary vehicles require new development; however, the electric cargo vehicle will require significant technical advances over SP-100

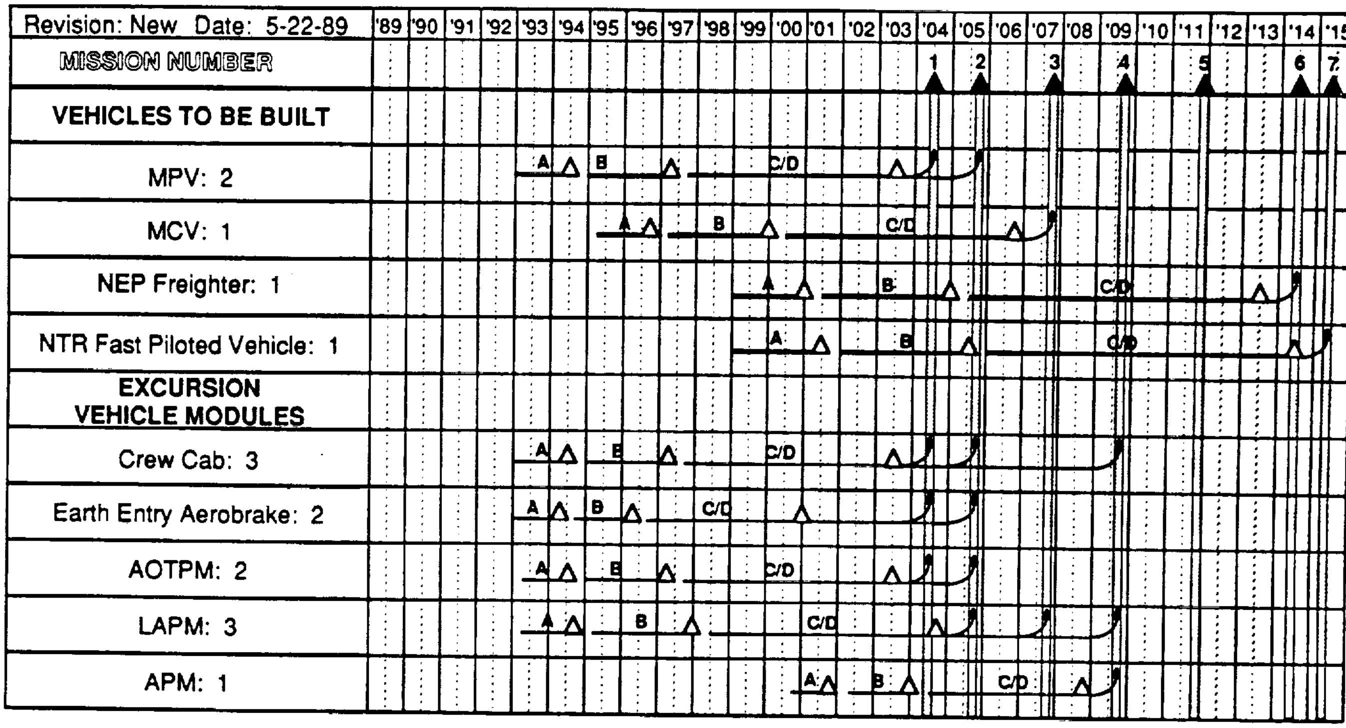


Figure 3.5-1 Vehicle and Module Development Schedule

class power systems to achieve 5 MWe power with a mass less than 100 tonnes.

Several technologies are required for a manned Mars mission regardless of specific objectives. First, on-orbit assembly capabilities, both robotically and using astronauts is required. This includes large structural deployments, cryogenic fluid connects, power and signal electrical connections, and visual inspection of self-deployed systems such as the large umbrella-shaped aerobrakes. Second, low boil-off TEI tankage is critical in controlling propellant mass requirements. Multilayer insulation, vapor cooled shields, thermodynamic hydrogen vents, and vacuum jackets will all be required. For the MDV and MCSV the hydrogen tanks will also require vacuum jackets and surface driven reliquefaction pumps. Third, Mars surface suits cannot be borrowed from current EVA or space station hard suits because of the combined requirements of low weight, flexibility, dust tolerance, and convective heat loss insulation. Forth, the TMI stage will require a new engine or highly modified SSME. If a new engine is to be built it must have a high expansion ratio, simple combustion and start-up cycle, and be highly reliable. Such an engine would have a thrust between 330 and 660 kN and an Isp of at least 465 seconds. If an SSME is to be modified then it must be fitted with a large nozzle, given a zero-g, space start capability, and preferably operate at 50% throttle level. Lastly, large flow-rate, zero-g propellant transfer capability must be developed for fueling the TMI stages. Several hundred to one thousand kilograms per hour is required.

Figure 3.6-1 shows the programmatic plan to develop these and other technologies. Some of the need dates are driven by the vehicle they are used in. In these cases the technology must be developed to level-7 6 months before the beginning of phase C/D for the vehicle employing the technology.

3.7 PRECURSOR NEEDS

The precursor needs for Mars Evolution are minimal because the initial mission delivers most of the satellites necessary at Mars to support the later missions. These include the Mars Rover Sample

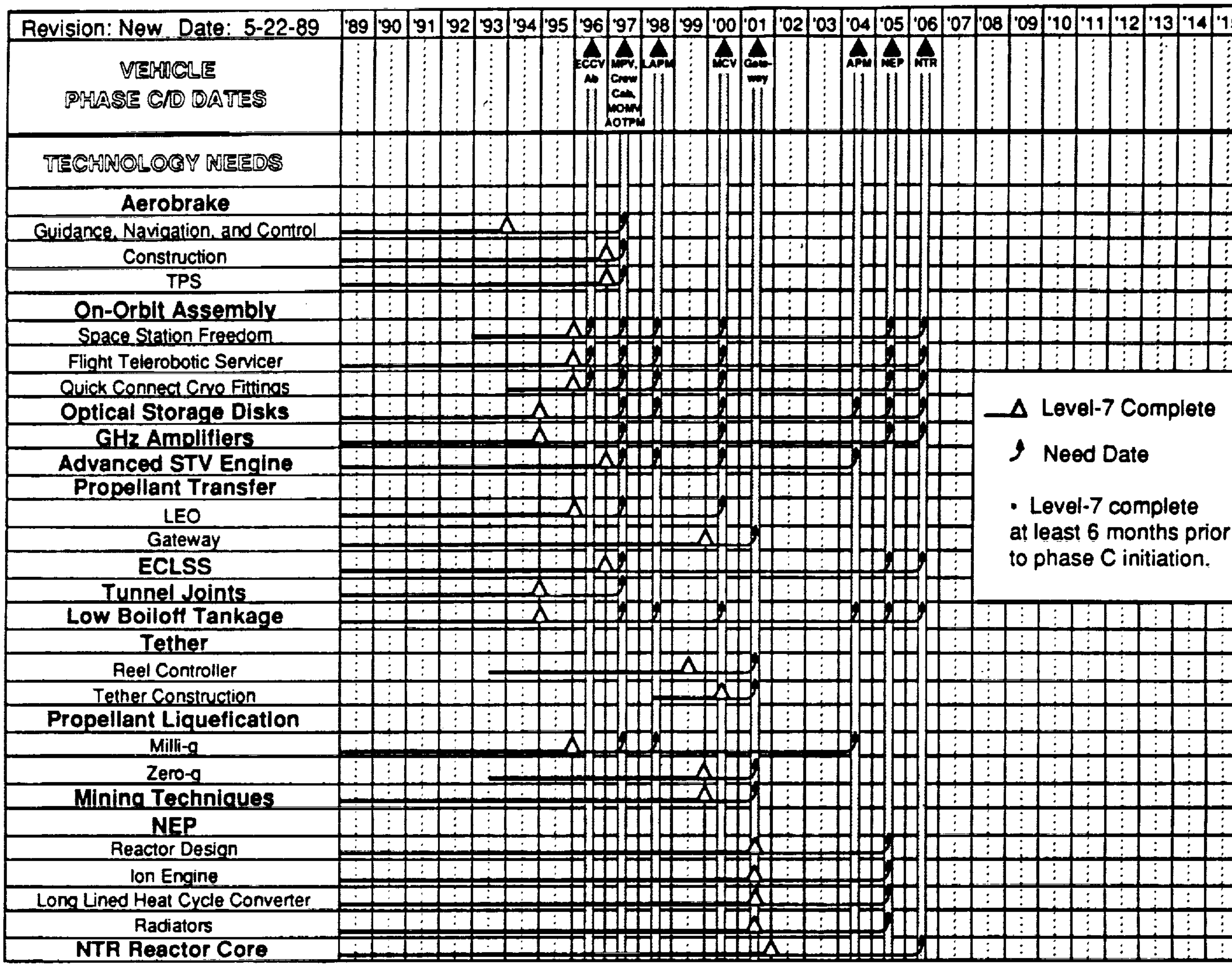


Figure 3.6-1 Technology Development Schedule

Return, 12 surface navigation beacons, two arian-synchronous relay/communications satellites, and a high resolution orbiting surface mapper. The only precursors needed fall into the class of system verification before committing humans to interplanetary flight. For example, an aerobraking demonstration of the flexible-fabric aerobrake design at Mars with worst case entry condition, or an Earth-orbiting MPV with crew on-board to demonstrate and refine the long-term suitability of the habitation module design, life support systems, and general human-machine interface.

3.8 HUMANS-IN-SPACE RESEARCH NEEDS

Humans-in-space research requires years of preparation, years of experiments, years of results analysis, and extended observations for long-term effects. The aggressive schedule for this case study (first mission in 2004) leaves too little time to fold these research results into the vehicle and operation plans. However, a parallel research activity could be initiated immediately that would influence downstream vehicle designs such as the NTR-PV.

Area's of research would study the long-term effects of exposure to the interplanetary radiation environment, tolerance of humans to high accelerations after several months of zero or reduced gravity, human's ability to adapt to constantly changing gravity levels, and finally, human's upper limit tolerances to spin rate, gravity gradient, and other rotational effects of artificial gravity.

3.9 TRADE RESULTS

Table 3.9-1 lists the ratio of the changed initial launch mass to the changed payload mass delivered to the indicated destination. One can see that 3 additional kilograms are needed for each kilogram delivered to a high elliptic orbit about Mars, Phobos, or brought back to Earth when Phobos propellants are used. If Phobos propellants are not available, as on missions-1 and 2 then the ratio increases to 3.6 to 4.6 respectively. Landing an extra kilogram on the surface of Mars only takes 4 kilograms in orbit, but, bringing it off the surface, back to Earth, requires an additional 18.3 kilograms. All these ratios include the payload mass itself in the increased mass figure.

Table 3.9-1 Sensitivity in IMLEO of Taking Payload to Different Destinations

Trip	"Gear Ratio"
Delivered to and Dropped at High Mars Orbit Mission-1: 18000 by 250 km Mission-2: 33120 by 250 km	 2.8 2.7
Landed onto Mars Using MDV (Mission-2)	4.0
Roundtrip to M. Surf and Return to Earth (Mission-2)	18.3
Roundtrip to Mars Orbit and Return to Earth Mission-1: 18000 by 250 km Mission-2: 33120 by 250 km	 4.6 3.6
Delivered to Phobos (Gateway) (i.e., MCV to MSurf)	2.8
Roundtrip to Phobos and Return to Earth Mission-4: Using Gateway Propellant Mission-5: Using Gateway Propellant	 2.9 2.9
Landed onto Mars Using MCSV (Mission-4) Roundtrip to M.Surf and Return to Earth (Mission-4)	2.9 2.9
Note: Phobos propellant usage zero's out all sensitivities after Phobos' rendezvous.	

3.9.1 TMIS Trade Study

TMI is a key candidate for trade studies because it makes up the majority of the initial mass parked in LEO–63 percent for the first mission. Several options are available for TMI: Firstly, a single Space Transfer Vehicle (STV) can be repeatedly refueled and sent to "push" on the MPV until it reaches escape, at which point the STV must have enough propellant to accelerate the MPV to the desired departure speed. Secondly, a stack of STV's could be integrated together, in series, parallel, or some combination of both. Lastly, a single, 400-tonne STV can be designed for Mars injection.

For this last option the oxygen can be accumulated in a holding tank until the majority of the stage's mass is in LEO. The oxygen tank is then attached to the MPV. Immediately before TMI, the entire load of hydrogen is launched in a single tank that also has the TMIS engines mounted on it. This tank/engine combination arrives in LEO, is mated to the oxygen tank and the Mars Vehicle and is ready for departure. This scenario eliminates the hydrogen boil-off problem, it eliminates multistage operations and hazards, and it keeps the most critical elements of the TMIS system on the Earth until the last possible moment.

For a multiple STV TMI scenario a trade study that addresses the following question was performed: How much additional mass and how many additional STV flights are required to be able to return one, two, or all of the STVs to LEO? In other words what is the marginal cost of making the STV stages used for TMI reusable?

The trade is performed with the single working STV scenario described above as option one. The maximum payload mass deliverable to Mars was first calculated using three fully loaded 140 tonne TMI stages that do not have aerobrakes and have a dry mass of 13 tonnes. This turned out to be 211.7 tonnes for a C_3 of 28.4 km^2/s^2. For any stages to be returned another stage must immediately be added. Referring to Figure 3.9.1-1 the three short bars

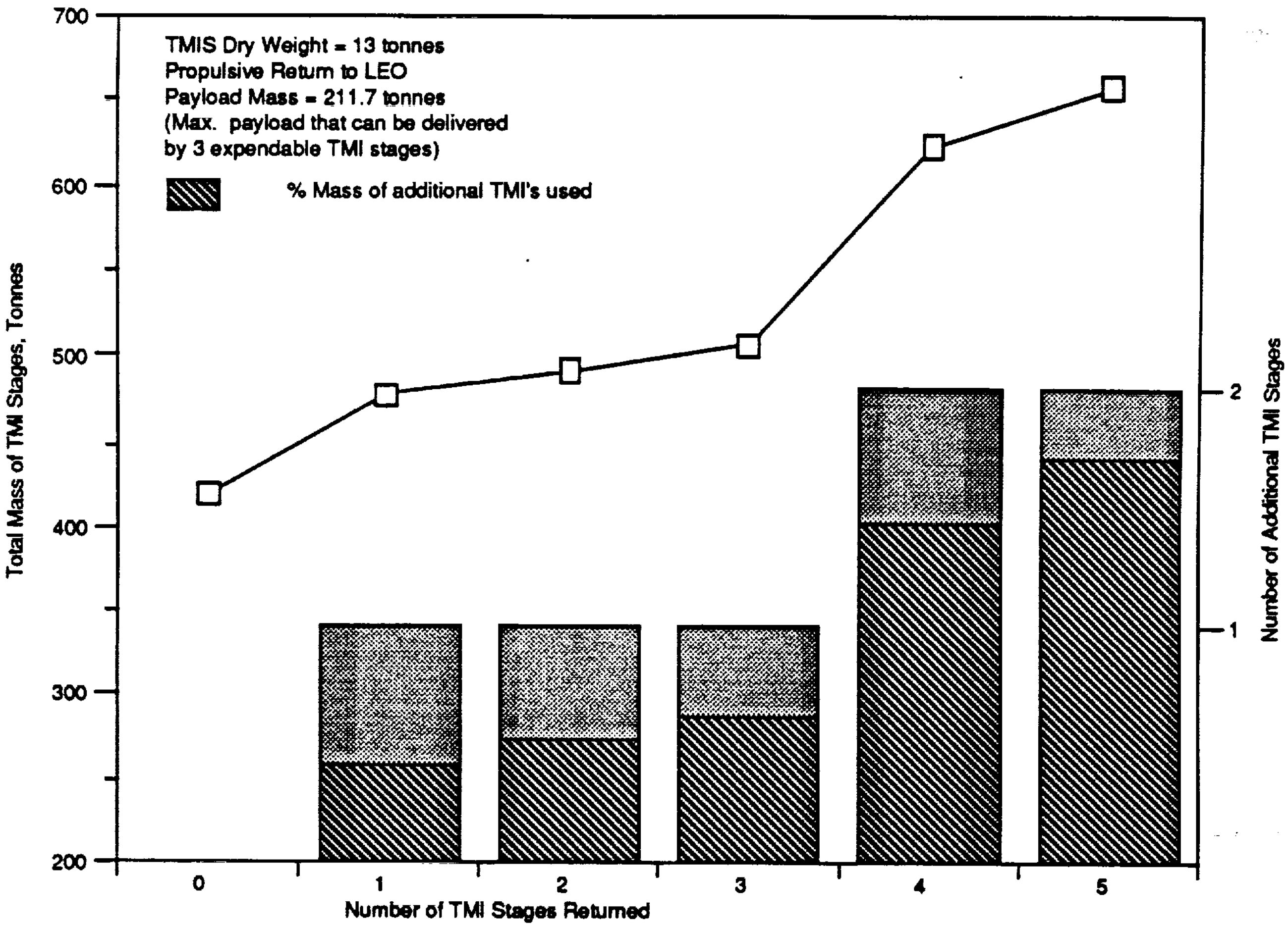


Figure 3.9.1-1 Marginal Cost of Additional TMI's for Payload Delivery

represent that added stage (right axis) with the amount of partial propellant loading shown within each (left axis). Hence, by adding a fourth STV the 211.7 tonne payload can be injected and up to three of the four STVs can be returned. This means the final STV is not returned. To return it, a fifth STV is needed to bring back the fourth and also to return itself. The line graph is the total TMI system mass as a function of how many STVs are returned to LEO after use. Note the jump associated with the addition of each STV to the scenario. In conclusion, the price for a fully reusable TMI system adds 80 percent more weight to the most massive element of a Mars mission.

The same trade study with the same ground rules was also performed using aerobraked STVs. Figure 3.9.1-2 shows that results are more favorable. Here, only one additional STV is required to have a fully reusable TMI system. The mass increase of 30 percent is more reasonable; however, these gains must be tempered by the increased STV dry mass that reduced the fully expendable payload capability of three STVs from 211.7 to 181 tonnes. Taking this into account the TMI system mass-to-payload mass ratio is 3.11 for the non-aerobraked STVs and 3.03 for the aerobraked STVs in the fully reusable scenarios. This compares to 1.98 and 2.32 respectively for the fully expendable scenarios. The ultimate driver in these trade studies is operability and cost. Hence, the cost of building and delivering a dry STV to LEO must ultimately be traded against the cost of delivering propellant to orbit.

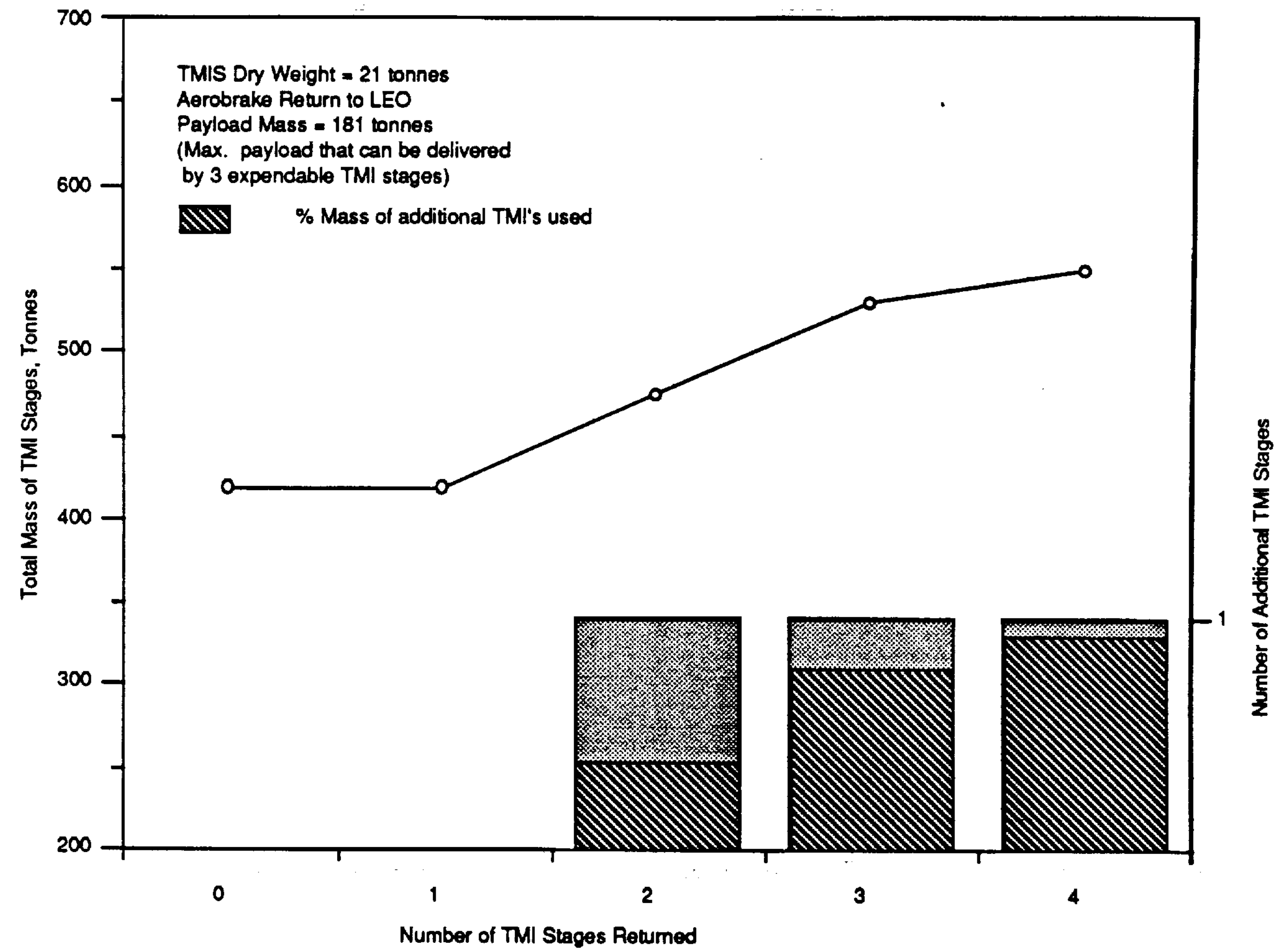


Figure 3.9.1-2 Marginal Cost of Additional TMI's for Payload Delivery

3.10 OPTIONS AND ALTERNATIVES

The complexity of Mars Evolution leaves ample room for options and alternatives. One option is to eliminate the Phobos tether. This is a reasonable alternative because the Phobos propellant production combined with the AOTPM can provide the same capabilities as the tether without increasing Earth-to-Mars mass flow. Another alternative is to add an additional cargo mission before mission-1 that delivers the infrastructure elements to Mars and Mars orbit. This mission could also carry an MCL that would leave a habitat on the surface. This would off-load the first two mission's cargo manifest and the MCV could be reused again on mission-3. Also, the unmanned flight of the 39-meter aerobrake carried on mission-0 would greatly increase the safety of the first piloted mission. Finally, it would greatly simplify the design requirements of the Phobos equipment if it could be assembled with the aid of humans. Hence, the mission following the gateway cargo mission (mission-3) should not plan to use the gateway facility, but instead, set them up.

3.11 CASE STUDY SUMMARY AND CONCLUSION

Mars Evolution is a complex case study because of the many vehicles and technologies specified by the requirements. The use of vehicle modules makes the scenario seem more complicated but in reality will reduce the overall program cost because of the high degree of module sharing. The vehicle modules also reduce initial mass in LEO by sharing the CCM between an excursion vehicle and the ECCV. Consideration was also given to module and component packaging for launch. Although the ETO limitations are 12.5-meters diameter by 25-meters length and 140 tonnes, the actual needed capability is only 10-meters diameter by 18-meters length and 80 tonnes.

One of the key objectives of this case study is to reduce the initial mass in LEO with the use of advance propulsion concepts. The tether system at Phobos is very effective in doing this. It can repay it's mass investment with the saved propellant mass of only two missions. The highest mass saving technology is the application of Phobos resources to make liquid hydrogen and oxygen. This capability, however, completely overlaps the advantage of the tether and should be staggered to come on line after the tether system has reached the end of its useful life. *In-situ* propellant production at Phobos is no trivial task. The low gravity environment dictates that special mining and surface operations methods be employed to prevent contaminating the entire area with floating debris. Dust kicked up by rocket engines could very rapidly create a permanent Phobos dust storm.

Mars Evolution will need further refining to assemble a realistic scenario. Currently the schedule is too tight in terms of launch frequency and time between now and the initial launch date. It has too many requirements for unique, follow-on vehicles that prematurely retire the existing vehicles—preventing them from being economically used. The mission sequence has overlapping missions that will require duplication of several ground facilities and space vehicles. It also puts a tremendous strain on the ETO schedule. It would make the scenario much more feasible if only one mission flew at a time and if all vehicles were used once in an unmanned mode before being committed to a piloted flight. This would dictate an unmanned precursor/cargo mission that would test all major elements of the MPV/MCV and ground control facilities.

The nature of these case studies is to rapidly assess several options for NASA to determine areas with and without merit. This case study is a valuable learning experience and with its strengths and weaknesses will lead to a more realistic, viable, and affordable sequence of Mars missions that will ultimately lead to the human colonization of another world. With these conclusions in hand, MASE further synthesized a derivative case study, as discussed in Section 5.2.2.

4.0 MARS EXPEDITION CASE STUDY (CS 2.1)

4.1 CASE STUDY OVERVIEW

4.1.1 Program Objectives

The Mars Expedition Case Study 2.1 has as the major thrust an early mission with the purpose of demonstrating the technological capability to send a small crew to the surface of Mars for a short exploratory visit and to safely return them to Earth. Total exposure time for the crew to the space environment is minimized. No infrastructure is emplaced at Mars, and the question of follow-on exploration and eventual settlement of Mars is not addressed in this case study.

4.1.2 Missions (Implementation)

The Mars Expedition Case study baselines a split-sprint mission, by which is meant that two separate trans-Mars vehicles are utilized, with the Mars cargo vehicle (MCV) following a high efficiency conjunction class trajectory and the Mars piloted vehicle (MPV) traveling on a less efficient trajectory which allows a much shorter round-trip time. The relatively small crew of three astronauts must rendezvous in Mars orbit with the cargo vehicle. In earlier split mission studies (Case Studies 1.0 and 2.0 of FY88), the return propellant or propulsion system — the trans-Earth Injection System (TEIS) — was part of the cargo mission. In the current case study, it has been baselined that the TEIS will travel with the crew and that the cargo mission consists of the Mars Descent Vehicle (MDV) and a relay communications satellite (RelayComSat). The latter is deployed automatically by the MCV upon Mars arrival in order to emplace the satellite into a proper orbit.

Although a full communications net in the case of a low Mars orbit for the piloted vehicle requires *two* RelayComSat's, it is a derived requirement that an appropriate precursor mission emplace one of the RelayComSat's. The first piloted mission is taken as 2002, with the precursor mission in 2001. However, vehicles are designed with propulsion system tanks to enable either the 2002 or the 2004 MPV launch opportunities.

4.1.3 Requirements

The case study requirements as given in the SRD provide specific guidance and/or constraints on vehicle design. Key requirements include the crew size of three, a 30-day staytime at Mars with 20 days on the surface, the need to land at altitudes as high as 5 km, and that aerocapture at Mars must be accomplished using a high lift-to-drag ratio aerobrake (L/D = 1.0). The HLLV system to support this mission is limited to four launches per year with a payload capacity of 140 t/launch and a limited shroud size of 12.5 m diameter. A full list of groundrules and requirements, referenced back to the SRD, and derived requirements are given in Appendix E. This includes also the deviations requested and the rationale for the request.

4.1.4 Assumptions

The calculation of mission mass allocations and the design of vehicle configurations is made possible by setting sufficient assumptions on technical approach. Some of these assumptions are determined by trade studies (see section 4.9), while others are more arbitrary. All are necessary to complete the analysis of the case study.

One of the most important assumptions is for direct entry. At Earth, the crew performs a direct entry and descent to the surface, with no provision for recapture of the interplanetary transfer vehicle. The module used to return the crew is entered just prior to encounter and is a small capsule, the Earth Crew Capture Vehicle (ECCV).

Another key assumption is adoption of the Mars Parking Orbit (MPO) as being circular, at 300 km, with specific inclinations depending upon the mission opportunity. This information results from analysis of orbital regression rates which, when combined with the orbital mechanics of Mars arrival and departure, is consistent with minimum propulsion requirements for accomplishing a short-duration mission at Mars. High elliptical orbits may also be satisfactory for certain of these mission opportunities and could lead to some reductions in IMLEO, but were not considered in this case study.

Other key assumptions include those concerning propulsion technology, propellant storage, and aerocapture technology. These topics are covered in more detail in following sections. Specific assumptions for conduct of this case study are given in Table 4.1.4-1.

4.2 VEHICLE DESCRIPTIONS

4.2.1 Configurations

The Mars Piloted Vehicle (MPV) in its aerobrake is shown in cross-section in Figure 4.2.1-1. The trans-Earth injection propellant is stowed in separate tanks aft of the two disk habitat modules, next to the six cryo-engines. In the nose of the L/D=1.0 aerobrake are located the pantry/radiation shelter and the Earth crew capture vehicle (ECCV). Tunnels connect the two disk module habitats and the pantry/rad shelter and ECCV modules are likewise connected to the pressurized habs to form a continuous shirt-sleeve environment. Trusswork, Figure 4.2.1-2 allows all modules and tanks to be combined into one integrated unit independent of the aerobrake structure. The aerobrake structure is shown in Figure 4.2.1-3.

The Mars Descent Vehicle (MDV) is contained in an identical high L/D biconic aerobrake, along with one relay communications satellite (RelayComSat), as shown in Figure 4.2.1-4, and with the Mars Ascent Vehicle (MAV) mounted on top of the MDV. Both vehicles are balanced to achieve the necessary location of the center of mass of the vehicle (see Appendix D) in order to maintain dynamic stability during the aerocapture event.

Table 4.1.4-1. Assumptions for Mars Expedition Case Study

- Direct entry at Earth (2.4.3.1.H).
  Note: Not a *requirement*. This appears only in 2.4.3, Ref. Mission.
  TIA accepts Direct Entry as baseline, however.
- No specific ΔV allocations for orbital launch windows (TMI, TEI) will be included in this Case Study
- Propulsion: Cryo H/O for TMI, TEI, DSM, MOO; storable biprop for MCC, MOC, RCS
  MAV is single-stage storable biprop.
- TMIS stages are 127 t propellant, 13 t dry (Multiple stages required)
- TMIS engines: SSME/HER per stage, 532 klb_f, I_{sp}=471 lb_f-s/lb_m (expansion ratio=300)
- TEIS engines: RL10-X1, I_{sp} = 470 lb_f-s/lb_m, 20 klb_f per engine
- MDV entry and landing: biprop for deorbit and terminal propulsion; aerobraking and parachutes
- Propellant margins (sum): 1% each for ΔV, I_{sp}, and bulk (except *no* bulk margin for SRD mandatory ΔV's)
  3% ΔV margin on MAV; 2% bulk margin on TEI
  Mandatory ΔV margins are used to size tanks only.
  Tanks are filled to correspond to the flight opportunity
- Hab modules: two 7.6 m dia [25-ft] disk modules, 2.7 m (9-ft) long
- Boiloff. Baseline is high boiloff for all cryogenic propulsion systems except TMIS
  Override trip times are included. All systems are fueled at T-3 months in LEO
  - low boiloff: 0.15 %/mo. LEO, 0.3%/mo. interplanetary (crew, sprint),
    0.1 %/mo. interplanetary (cargo, Cn), 0.065%/mo. at Mars
  - med boiloff: 0.33 %/mo. LEO, 0.6 %/mo. interplanetary (crew, sprint),
    0.2 %/mo. interplanetary (cargo, Cn), 0.15 %/mo at Mars
  - high boiloff: 0.55 %/mo. LEO, 1.0 %/mo. interplanetary (crew, sprint),
    0.4 %/mo. interplanetary (cargo, Cn), 0.33 %/mo at Mars
  TMIS boiloff is 3.0%/mo. in LEO. Average time in LEO is 3 months.
- PVPA for spaceborne power, 100 m² (13 kW_e at 1 A.U.; 7 kW_e at worst-case Mars, 4.5 kW_e in LMO)
- Spaceborne ECLSS: closed for H_2O, O_2, CO_2; open 100% for food; open for make-up atmosphere
- Pressurized atmosphere: 10.2 psi (31 % O_2, 69 % N_2)
  Leakage rate: 0.03%/day. Cabin ventings: 5 per mission
  All hardware qualified for 5.0 psi (69 % O_2, 31 % N_2,) for emergency operation
- Voice-activated emergency command and control
- Aerocapture technology: very conservative (biconic brake is 20% mass surcharge; low L/D brake is 15%)
- Aerocapture brake is rigid for nominal case. Aerobrakes are foldable, flex-fabric for low L/D alternative
- MDV aerobrake is 5% mass fraction, ballistic coefficient of 100 kg/m²
- MDV habitat: one 4.5 m [15-ft] diameter disk module
- Landed ECLSS: no O_2, CO_2, or water recycling
- MAV cargo is 150 kg of samples taken or exposed at the Martian surface; 100 kg returned to Earth
- Mars Parking Orbit (MPO) is at 300 km circular.
  Inclination = 50° for 2002 mission, i = 20° for 2004 mission

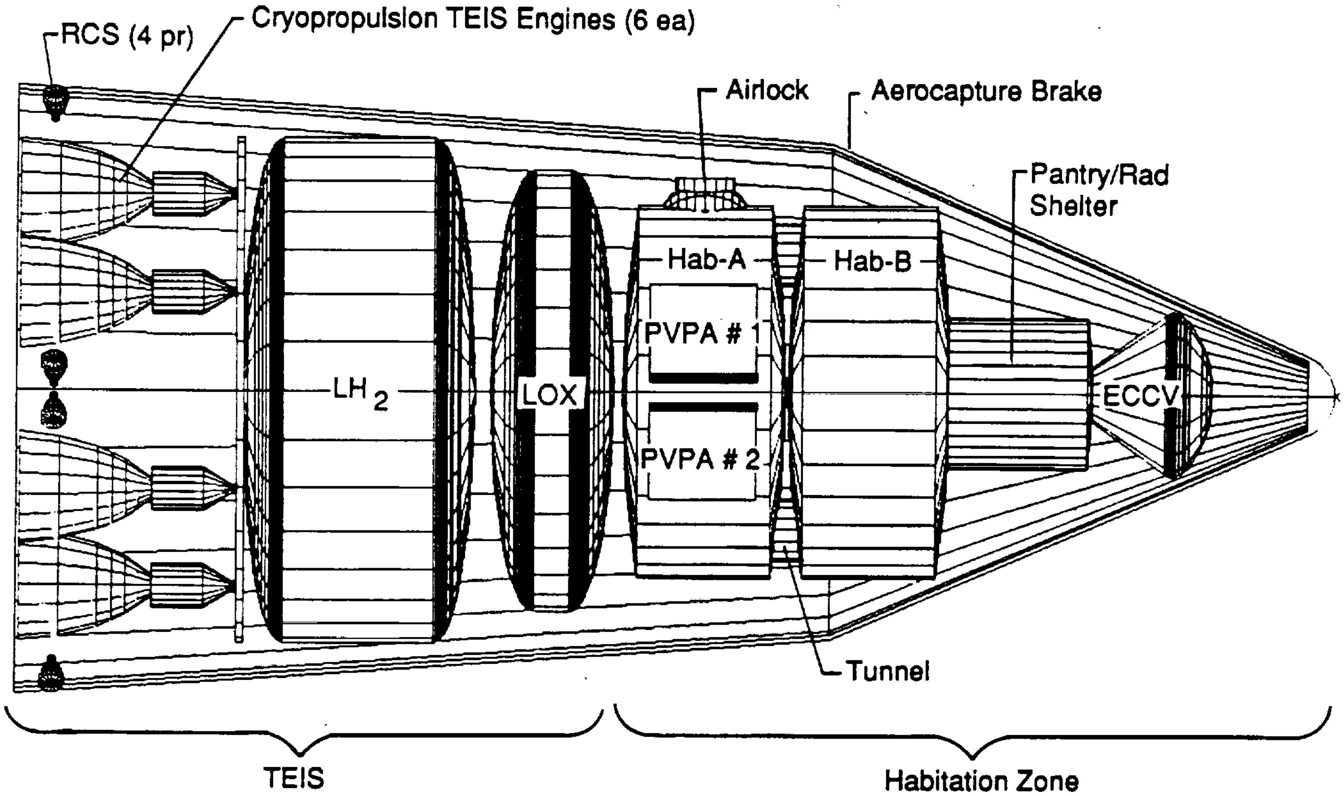


Figure 4.2.1-1 Mars Piloted Vehicle

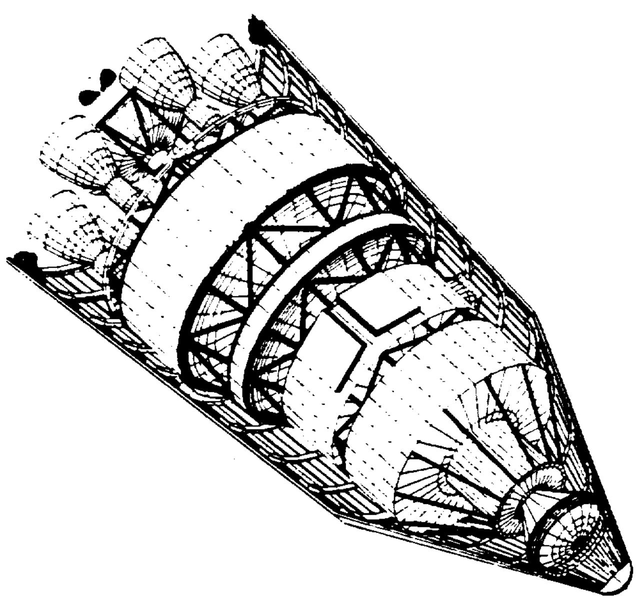


Figure 4.2.1-2 MPV Interstructure Trusswork

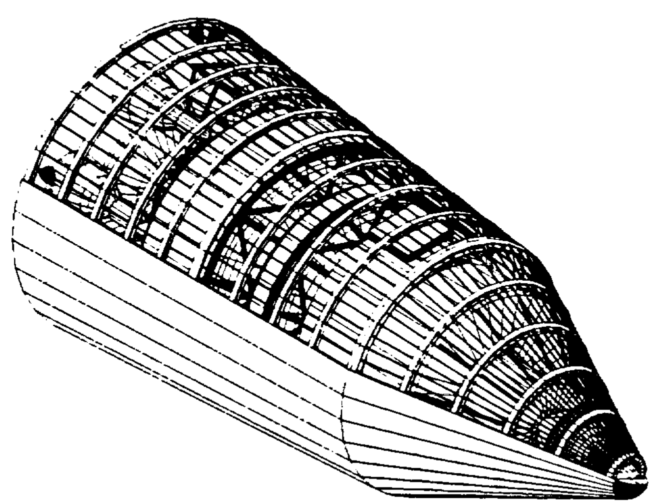


Figure 4.2.1-3 MPV Aerobrake Design

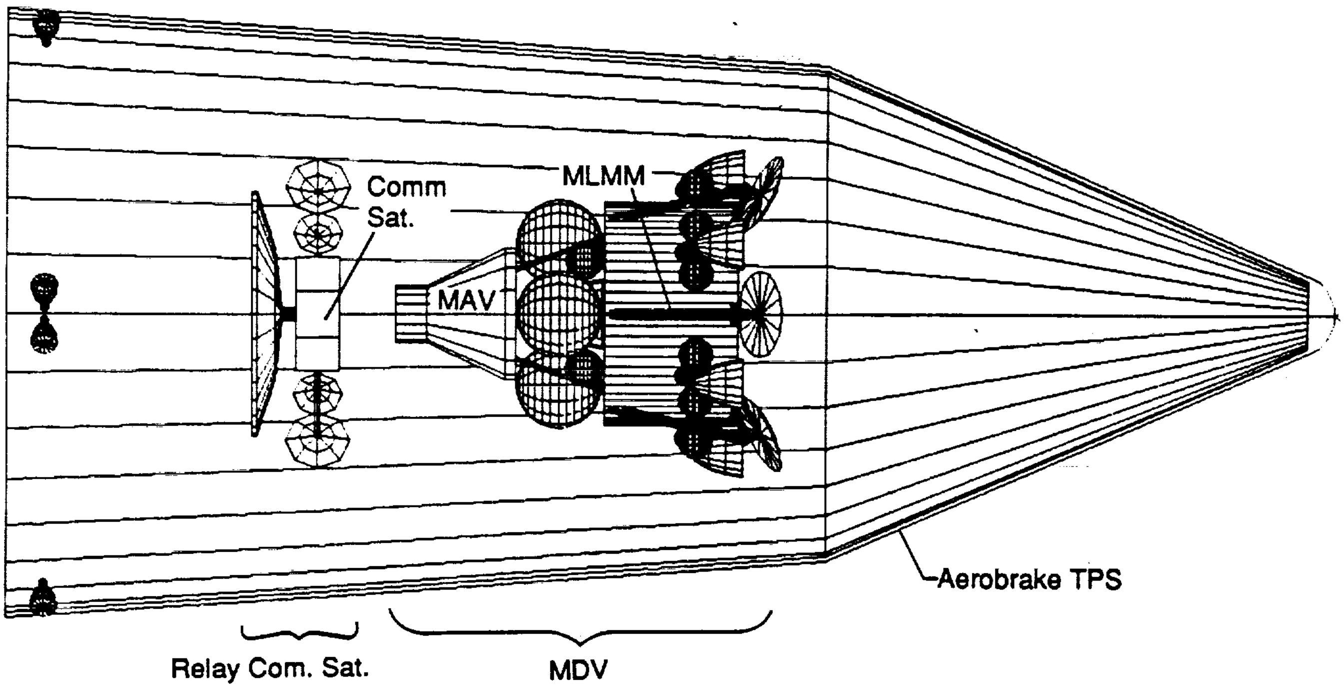


Figure 4.2.1-4 Mars Cargo Vehicle

4.2.2 Element Summaries

The MPV is a highly integrated unit because of the complex interrelationship of structural support and center of mass location with respect to the brake exoskeleton. The outer envelope, i.e., the aerobrake skinline, is 12.5-m in maximum diameter, 27.1-m long, with a 23.5° nose angle. The Earth return vehicle, which consists only of the hab modules, pantry/rad shelter, and ECCV has a gross mass of 39.5 t. The aerobrake mass is 31.0 t, which is about 18.6% of the mass being braked into orbit at Mars. The MDV and MAV are described in Figures 4.2.2-1 and -2, respectively. The RelayComSat is allocated a mass of 2.0 t for the satellite and its propulsion system to reach its assigned Mars parking orbit.

4.2.3 Commonality

In order to achieve maximum commonality in the aerobrake, they were made identical for the two flights. The piloted mission is the actual driver for aerobrake size and weight, both because of its higher mission mass and because of the trajectory, which enters the martian atmosphere at a higher velocity than the cargo mission.

4.2.4 Cargo Accommodation

No pallet-style cargo hold is provided for this mission. However, the MDV landed mission module is a 4.5-m diameter by 3-m high disk habitat which contains ample storage in the false floor, the false ceiling, within the interior, and also exterior to the module. A total of 10 t is accommodated.

4.2.5 Science Accommodation

Any split can be made as desired in the cargo mission between science equipment and other equipment for delivery to the martian surface. Volume accommodations range from exterior unprotected environment to interior fully conditioned environment. In addition, 450 kg of interplanetary science equipment is allocated to the piloted vehicle. A portion of this must be solar monitoring equipment to provide advance warning

Payload Mass	
(includes MLMM and Equipment)	10,000 kg
Payload Volume	
MAV	
(cone - 2.4m dia., 2.4m ht.)	6.3 m^3
HAB Module	
(cyl.- 4.6m dia., 2.7m ht.)	44.9 m^3
Propulsion System, Descent	
Propellant Type	MMH/N_2O_4
Engines	
Number	6
Type	Shuttle-OMS
Mass (ea.)	134 kg
Thrust (total)	160 kN (36 klb_f)
I_{sp} (316 sec)	3.1 kN-s/kg
Propellant Mass	6,660 kg
Tank Mass	200 kg
Terminal T/W	0.55 gee
Total Mass	35,200 kg

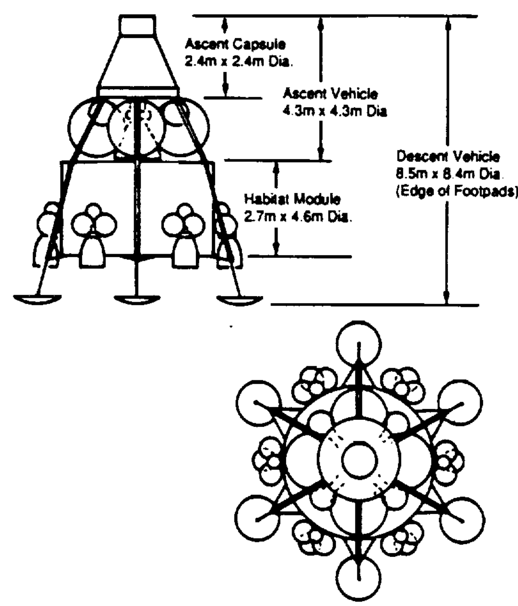


Figure 4.2.2-1 Mars Descent Vehicle (MDV)

Payload Mass to LMO	2,100 kg
Payload Volume	
(cyl. - 2.4m dia., 2.4m ht.)	3.6 m^3
Propulsion System	
Propellant Type	A-50/N_2O_4
Engines	
Number	1
Type	Apollo-CSM
Mass (ea.)	373 kg
Thrust (total)	91.3 kN (20.5 klb_f)
I_{sp} (314 sec)	3.08 kN-s/kg
Propellant Mass	8,690 kg
Tank Mass	340 kg
Initial T/W	0.69 gee
Total Mass	11,000 kg

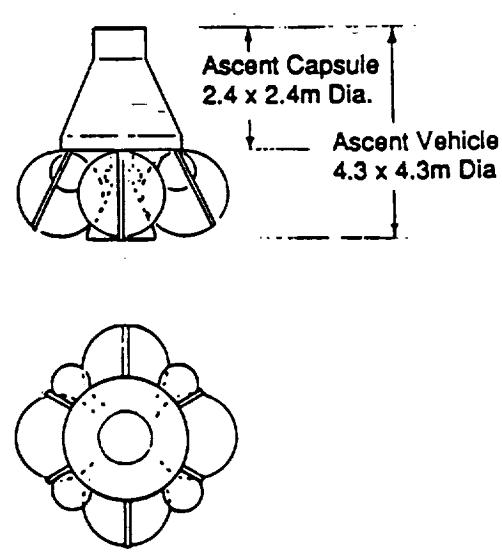


Figure 4.2.2-2 Mars Ascent Vehicle (MAV)

of potentially hazardous solar flare activity. An additional 150 kg of Mars orbital science equipment is also provided. A key trade study has demonstrated that eliminating all science whatsoever results in only a 2% reduction in IMLEO. Conversely, for a very high science payload (14,050 kg increase, including MRSR modules, Ph/D teleoperator, Venus probes, two manned and two unmanned rovers, and a Mars science satellite), the IMLEO increases by only 45.6 t (about 5.5%). Thus, a human mission to Mars can accommodate large amounts of science equipment without seriously effecting the mass to LEO.

4.3 SYSTEM DESCRIPTIONS

4.3.1 Habitats

The living space for the three crew members consists of two separate, somewhat redundant modules in the shape of squat cylinders (called "disk modules"). The purpose of having two habitats is to provide a capability for rapid exit to a safe haven in the event of sudden, major systems problem with one of the habitats. For example, a large micrometeoroid could breach the pressure vessel, a spot fire or contamination spill could occur, or a power system could be accidentally crossed to ground, necessitating interruption of power to locate the fault. In addition, two modules provides some psychological relief for the extremely isolated and confined environment (ICE). Two tunnels are likewise provided to provide for dual egress from all hab modules at any given time. All tunnels and entryways have dual pressure-isolation hatches, if needed. The fore-most module actually has triple egress, counting the pantry and radiation shelter module. This latter module contains much of the food supplies and ECLSS equipment consumables, located in wall lockers to provide a densely shielded region for refuge during major solar flare particle events (SPEs), as shown in Figure 4.3.1-1. Up to 35 g/cm^2 of wall-thickness shielding can be maintained if removed consumables are replaced by waste materials during the course of the mission. Detailed calculations show that this shield can reduce even the February 1956 relativistic SPE and the high-flux November 1960 SPE to doses of less than 15 rem to the blood-forming organs of the astronauts.

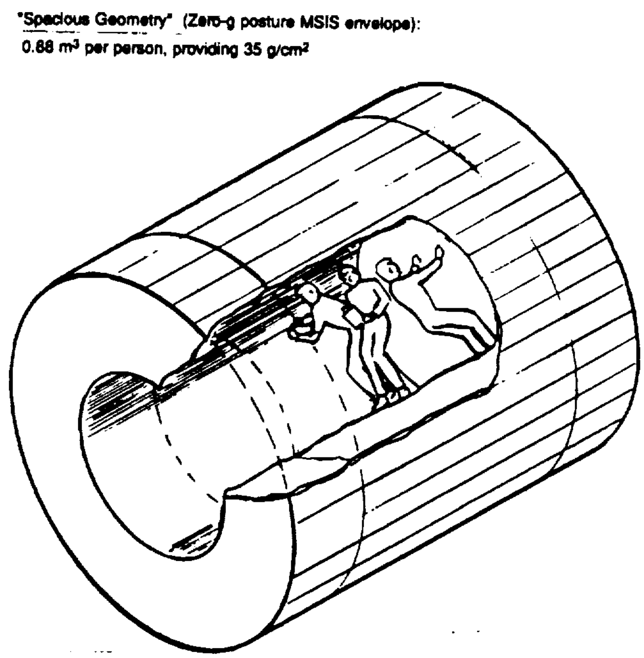


Figure 4.3.1-1 Pantry/Rad Shelter

Inside the main habitat disk modules, a zero-g living environment is arranged, as shown in Figure 4.3.1-2, and includes individual personal quarters for each crewmember, as well as compartmentalized personal hygiene area, fitness maintenance

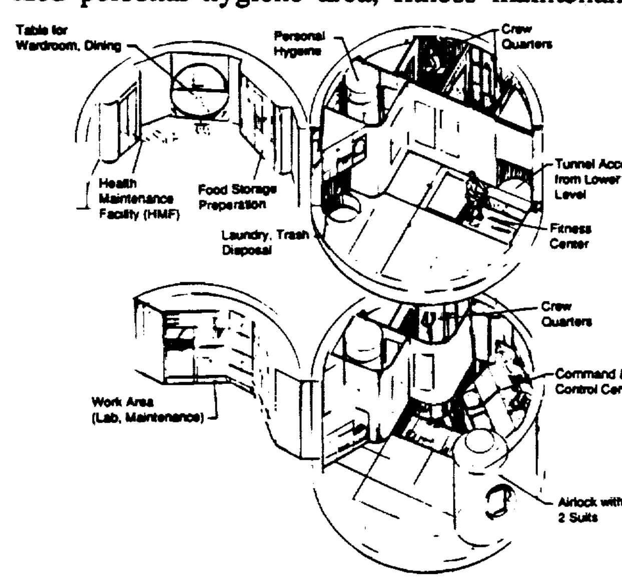


Figure 4.3.1-2 Habitat Interiors

center, inset food storage and preparation zone, mess table, and the work and command and control instrumentation centers. Note that this volume is probably the minimum acceptable for such a long voyage, and was chosen for this case study only because of the groundrule to seek a minimum IMLEO for an early and purely expeditionary mission to Mars. More details of the crew quarters are evident in Figure 4.3.1-3. This is a highly functional crew accommodation, but nonetheless an ICE.

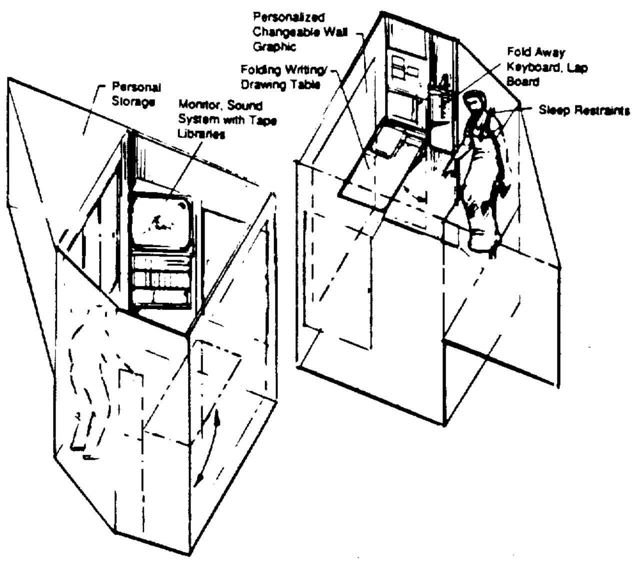


Figure 4.3.1-3 Details of Crew Quarters

The ECLSS for this mission will consist of a highly closed, recyling physical-chemical system. Water is reclaimed from hygiene waste waters as well as urine, at least early in the mission. Oxygen is reclaimed from carbon dioxide. The food source is fully open, however, and there are no provisions for growing of crops in this early mission. In any event, because of potential complications in the closed ecological system of such an environment, biological organisms such as plants and animals may be barred from the voyage.

4.3.2 Propulsion Systems

The trans-Mars injection system (TMIS) for these flights is planned to be derived from high energy upper stages assumed to be part of a new HLLV launcher. The TMIS is comprised of three stages, one of which is shown in Figure 4.3.2-1. Each stage can accommodate up to 127 t of liquid hydrogen/liquid oxygen (H/O) propellant. Three stages are needed for the MPV, with the first stage off-loaded by less than 1 tonne. The MCV requires only a single stage, and it is filled to only 119 of its 127 t capacity. The engine assumed for this major upper stage has a performance of 4.61 kN-s/kg [471 lb_f-s/lb_m], consistent with a vacuum-qualified SSME having an extendable nozzle to achieve the expansion ratio of 306:1.

The trans-Earth injection system is also H/O based, and is designed around a sextuplet of engines derived from the Centaur RL-10 (the -X1, with I_{sp} = 470 lb_f-s/lb_m and thrust of 20 klb_f per engine). With the large engine cluster, dual fault tolerance is enabled (two engine-out capability), and up to 3 or even 4 engines could be off and still allow Mars escape on an Earth-bound trajectory.

It is assumed that the TEIS is "high" boiloff (see Table 2.1.4-1) in order to be conservative and to acknowledge the difficulty of achieving the requisite high thermal isolation between the proximate habitat modules and the cryogen propellants. This boiloff rate is accomplished by mainly passive means, including vapor cooled shields, superinsulation, and a thermodynamic vent. As will become apparent in the portraying of flight configurations, this system has a large view factor to deep space during all phases except aerocapture, LEO operations, and in Mars orbit. If a high elliptical orbit is selected at Mars rather than the nominal 300 km circular orbit, the martian albedo and infrared emission fluxes should be of little or no impact because of the favorableness of the time-distance relationship. A total of 92.2 t of TEI propellant is needed for the return to Earth.

For deorbit and terminal descent on the MDV, an array of six storable bipropellant engines and tank sets are provided. These engines are placed as far outboard as possible to minimize blast mobilization of soil and to provide attitude control under strong

Dry Mass	13,000 kg
Payload Mass	variable
Propulsion System	
Propellant Type	LOX/LH2
Propellant Capacity	127,000 kg
Tank	
Mass	8,890 kg
Tankage Factor	7.0 %
Length (40 ft.)	12.2 m
Diameter (24.0 ft.)	7.3 m
Engines	
Number	1
Type	SSME/HER
Mass	3,710 kg
Size	
Length, extended (25 ft.)	7.6 m
Length, nested (14.4 ft.)	4.4 m
Exit Diameter (15 ft.)	4.6 m
Thrust @ 100% (532 klbf)	2366 kN
Isp (470 s)	4.61 kN-s/kg
Expansion Ratio	306:1
Total Length, nested (58.2 ft.)	17.8 m
, extended (68.7 ft.)	21.0 m
Total Mass (Wet)	140,000 kg

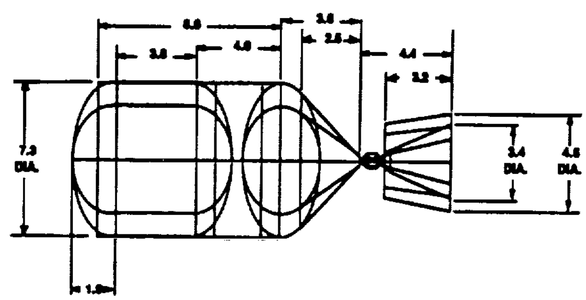


Figure 4.3.2-1 TMIS Propulsion Module

wind conditions. Shuttle OMS engines (I_{sp} = 316 lb_f-s/lb_m and thrust of 6 klb_f) are used in this application. The Mars Ascent Vehicle (MAV) is powered by a single storable bipropellant engine of the same class as used in the Apollo Service Module propulsion system (I_{sp} = 314 lb_f-s/lb_m and thrust of 20.5 klb_f)

4.3.3 Aeroassist Systems

The Mars aerocapture brake was shown in Figure 4.2.1-3. It is an all-rigid system using hard insulator tiles and heat-resistant, advanced composite support structure, with an L/D of 1.0. The MDV entry aerobrake is a hybrid of a hard inner core and flexible fabric annulus of TABI (see section 2.3.3, Lunar Evolution) to allow on-orbit automatic deployment and avoid assembly in LEO. It has an L/D of 0.25. The ECCV employs an ablator aeroshield, is Apollo-shaped, and has an L/D of 0.3.

4.3.4 Communication Systems

The driving factor for communications bandwidth is the need for near-continuous video channels. It is recommended that two downlink channels be in effect at any one time, even though the spacecraft may have up to a dozen or more active cameras at strategic locations. One of these channels would be controlled by the on-board crew, the other by the Earth-based monitoring team. This allows flexibility on the part of each group. Each day there could be high-power transmission of up to six video channels for approximately 2.4 hrs total. In addition, a number of low-rate video channels would perform housekeeping and solar monitoring functions. For other data, 420 engineering channels at 35 kbps total and an allocation of 160 kbps for science and solar patrol are provided. Uplink to the spacecraft is similar, but with data and computer software uplink in place of data downlink. Communications with the lander are provided by strategic placement of one or two RelayComSat's. If the 250 km x 1 sol orbit is selected for the Mars Parking Orbit (MPO), then a single RelayComSat placed in mirror image around the planet can provide the desired link. Additional information on data rate allocation was given in Tables 3.3.4-1 and -2.

4.3.5 Power Systems

The deployable-retractable photovoltaic power arrays (PVPA; solar cell panels) provide a minimum of 100 m², which will produce 13 kW_e at 1 A.U. from the sun and approximately 7 kW_e for the perihelion case at the distance of Mars' orbit. If the MPV is in a low Mars orbit, solar occultations could reduce this value to as low as 4.5 kW_e. Similarly, in LEO, the average power is expected to be less than 8 kW_e. Note that there is no need for nuclear power in this mission. It is assumed that the ECLSS system, especially the power-intensive ventilation and heat dissipation subsystem, will be of lower power than for SSF by incorporation of modestly advanced technological approaches. Since no high power load experiments, such as the microgravity materials processing research investigations on SSF, would be appropriate for the manned Mars mission, it is assumed that this is not a concern.

4.3.6 Thermal Systems

The in-space thermal control system can be mainly passive or utilize a single liquid loop into radiators hidden behind the PVPAs. This is because of the relatively low heat generation on the vehicle.

The thermal control system for the aerocapture pass consists of passive preventative measures in the design of the aerobrake and its interface with the habitats and cryogens. More information is provided with respect to the aeroassist discussion, section 4.3.3.

4.4 OPERATIONS CONCEPT

4.4.1 ETO Manifest

The Earth-to-orbit launch sequence is shown in Figure 4.4.1-1, derived from the baselined capability of 140 t to orbit per launch. A Shuttle-Z approach is assumed, whereby the upper stage of the HLLV can be reused on-orbit to serve as a stage in the TMI system. In panel (a) of this figure, the MCV is shown being launched on the first HLLV. Total payload is only 77.6 t, followed by the (b) launch which is to carry the single TMIS stage and its propellant.

The MPV is launched partially dry (without TEIS propellant). The gross payload mass is 105.1 t for the (c) launch. Note that in both the (a) and (c) cases, the HLLV launch shroud can be eliminated because the aerobrakes can serve this purpose. In (d), 92.2 t of TEIS cryopropellant are lifted. Only the (e) through (g) launches remain, to deploy the three TMIS stages and their propellant loads. Launch (h) is for crew transfer to orbit, but there could also have been additional STS launches prior to this time for on-orbit inspection and checkout of systems.

4.4.2 On-orbit Assembly

No on-orbit assembly of the MPV is required for the Mars Expedition design because the entire vehicle is launched all-up. However, there will be de-docking of the vehicle from its aerobrake. One major reason for this strategy is to facilitate TEIS propellant transfer. Upon reaching orbit, a TMIS stage must receive its propellant from the payload tank by automatic in-space transfer. Since the systems is launched with plumbing in place and propulsive settling is available, this transfer is straightforward. Automatic disconnects permit jettison of the payload tank after the transfer is completed. Docking of TMIS stages is another key on-orbit maneuver. Because these stages are totally independent from one another, no fluid or electrical interconnects are required (although a hard line for commands may be desirable for sequencing igni-

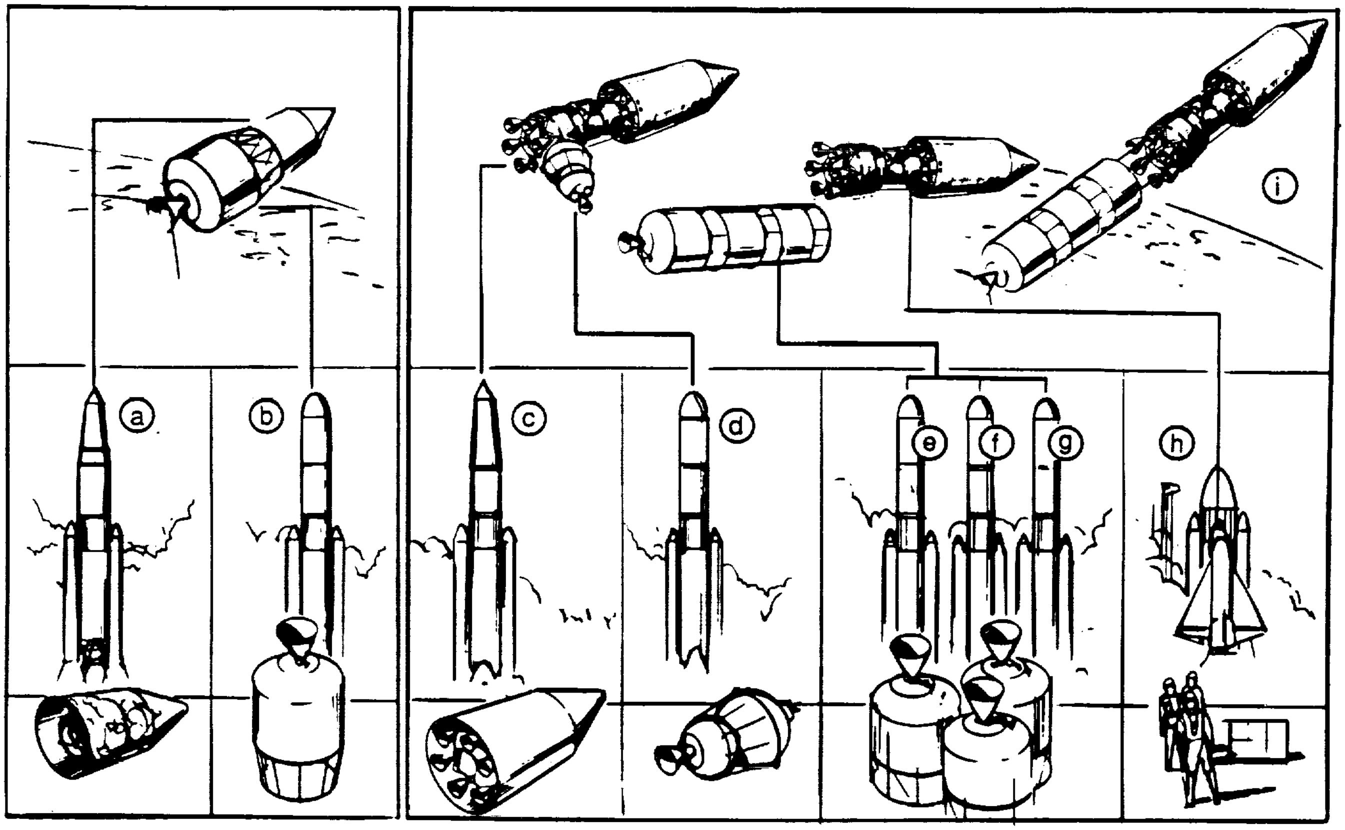


Figure 4.4.1-1 ETO Sequence

tions and staging) and the docking is purely an aligned latching. The stages are "flown" from the ground, using teleoperated command and control.

4.4.3 Mission Operations Sequences

As portrayed in Figure 4.4.1-1, the cargo mission requires only two launches because only a single TMIS stage is needed. The piloted vehicle is launched on its trans-Mars trajectory by three successive burns of the large TMIS stages. The vehicle is kept rolled out of its aerobrake, but with PVPAs retracted for the burn.

A typical flight path, the 2004 piloted mission, is shown in Figure 4.4.3-1. At the Venus swingby, the crew could deploy scientific probes. Special thermal protection devices, such as sunshades, may be deployed to aid in thermal control during the leg inward through the solar system. At Mars, the MCV, arriving first, accomplishes aerocapture and then deployment of the RelayComSat, panel (k) of Figure 4.4.3-2. The MDV and its aerobrake then await in MPO for the arrival of the MPV. As the

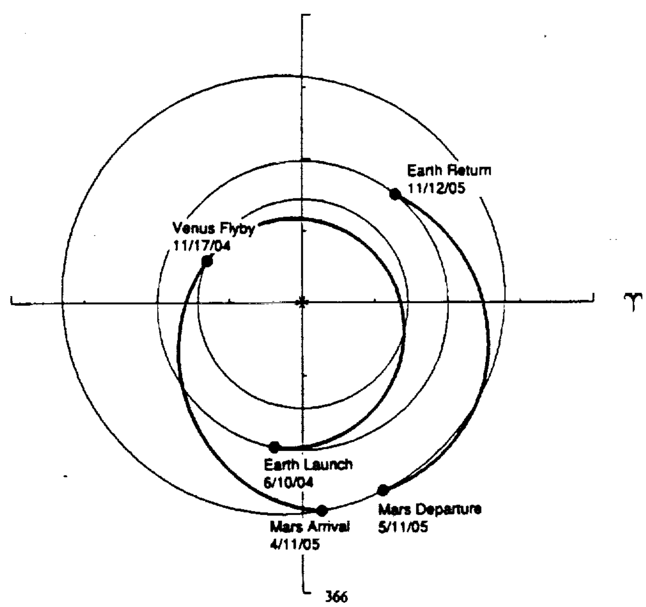


Figure 4.4.3-1 Mars Expedition - 2004 Launch

MPV approaches Mars, the vehicle must re-dock into the aerobrake. During the entire interplanetary flight, the two were in the extended configuration because of the many advantages of the "out-of-the-brake" configuration: better thermal management (dissipation of manned systems waste heat and view factor for cryogenic systems to radiate to deep space); deployment of PVPAs and radiators; astrophysical science measurements and solar monitoring; simplified egress for astronauts and accompanying free-flying robot(s).

Upon entering the aerobrake, the MPV switches to a special low power mode, drawing from batteries and fuel cell power supplies incorporated with the TEIS. The LSS is put into a non-recycling mode to reduce power drain and heat generation. In conjunction with test burns of the reaction control system (RCS) to determine the cg location, and the redistribution of moveable supplies, the cg is trimmed for entry. Terminal navigation is provided by radio links with the RelayComSat's and any other available navigation aids, such as precursor satellites and landed beacons. During the aerocapture pass through the martian atmosphere, the MPV is controlled automatically to compensate for atmospheric density variations and initial targeting errors. Immediately after the aeropass, a burn of the TEIS raises periapsis to circularize the orbit at the desired 300 km. The MPV then disengages from its aerobrake, which it no longer needs. PVPAs are deployed and the MPV begins its rendezvous sequence with the predeployed MCV.

Upon achieving rendezvous, the crew transfers from the MPV to the MDV. There are several methods by which this could be accomplished, including MMU/EVA fly-over or docking. In panel (n) of Figure 4.4.3-2, the MDV is shown as having exited its aerobrake and docked with the aft hab module of the MPV. In this scenario, the MDV then re-engages inside its aerobrake, panel (o), before performing the deorbit burn and entry, panel (p).

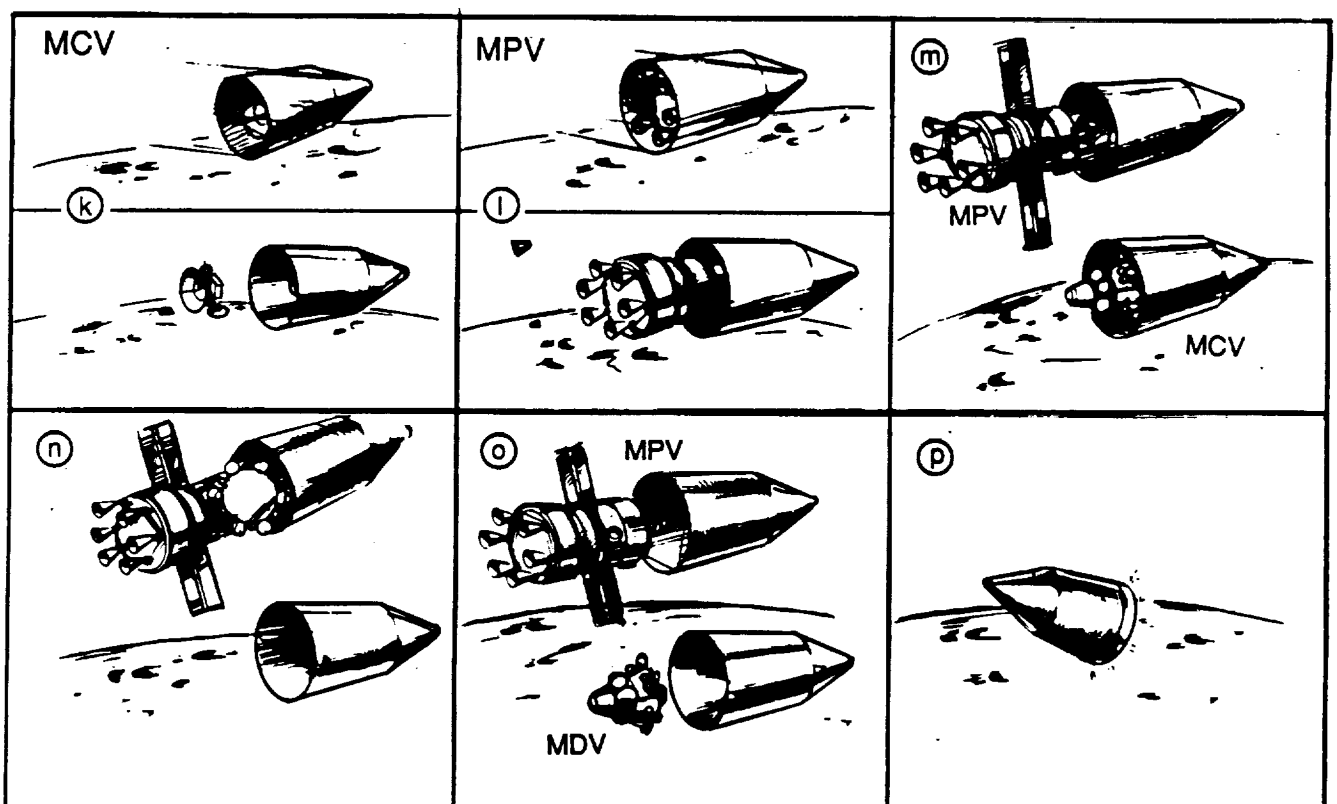


Figure 4.4.3-2 Mars Vicinity Sequence

The descent sequence is schematically depicted in Figure 4.4.3-3. During the terminal descent phase, the MDV can accomplish cross-range translation to achieve a pinpoint landing and/or to avoid hazardous landing areas. Use of the parachute depends upon the altitude of the landing site. A requirement for landing at up to +5 km altitude should be relaxed to a lower value to permit implementation of parachute-assisted deceleration (large areas of the martian surface are below +3 km altitude).

Upon landing, the astronauts disembark to deploy science equipment and supplemental power sources, including a surface solar cell array. Twenty days later, and after a direct transfer into orbit via an ascent flight of about 10 minutes, the astronauts accomplish rendezvous with the unoccupied MPV. After docking and shirtsleeve transfer into the MPV, the MAV is discarded. Thirty days after aerocapturing at Mars, the return-to-Earth is initiated by a TEIS burn sequence. Shortly prior to encounter with Earth, the crew enters the ECCV and releases from the main MPV. A direct entry and splashdown, in the Apollo command module style, completes the mission (see Figure 4.4.3-4).

4.4.4 Reliability and Safety

Assurance of the safety of the astronauts and maximization of the probability of total mission success requires vigilant attention to detailed implementations as well as a thorough and properly conceived management plan. This will be accomplished in the proven manner NASA has developed over the years in numerous manned flight programs. Important components of a Mission Assurance program for the long-duration Mars missions are delineated in Table 4.4.4-1.

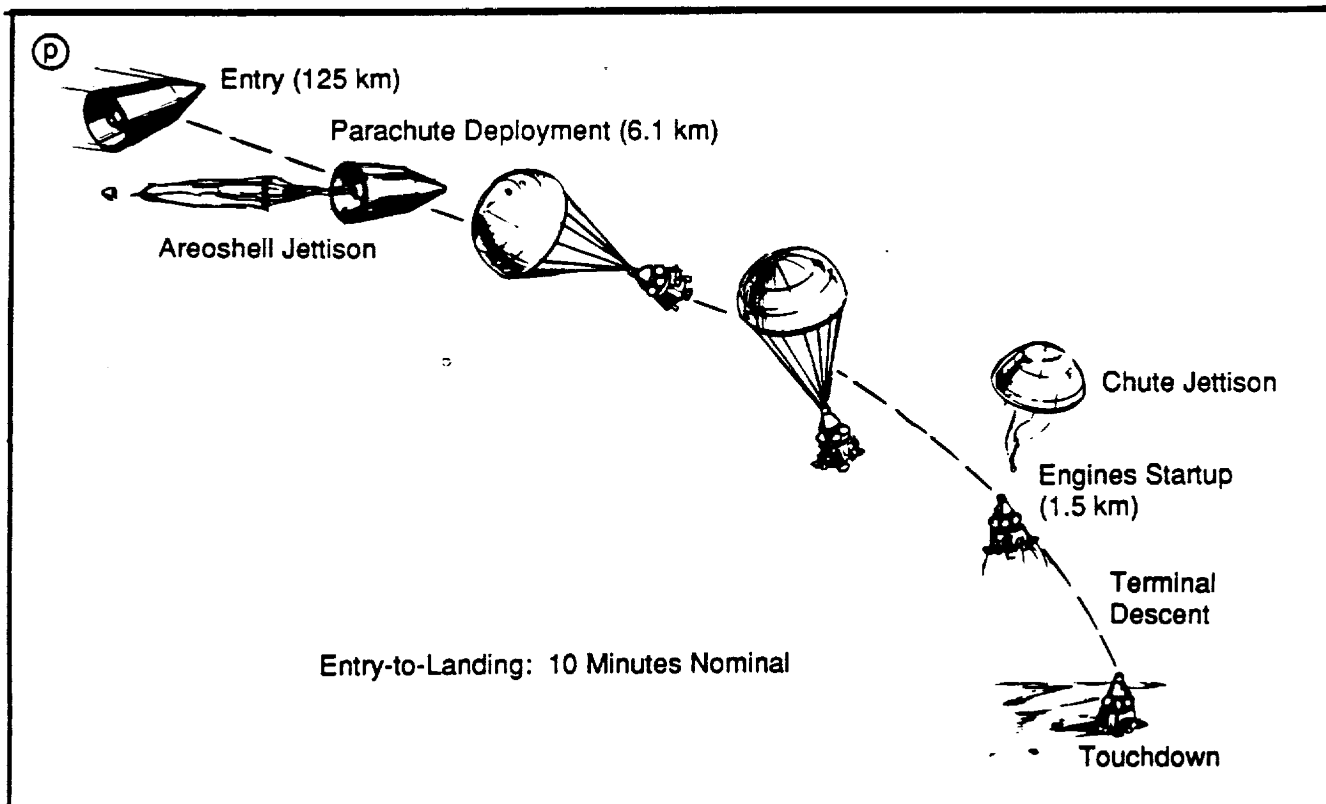


Figure 4.4.3-3 Mars Entry and Landing System (MELS) Sequence

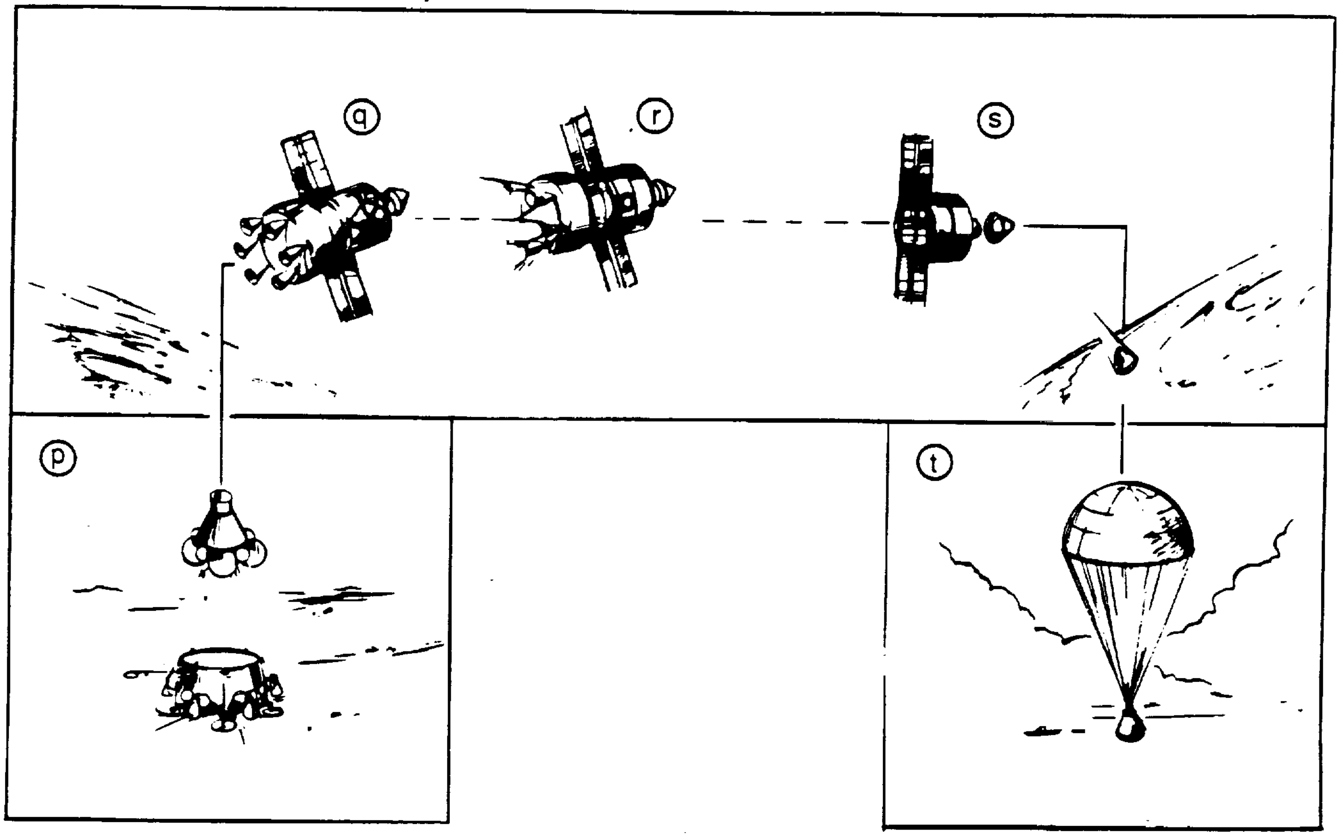


Figure 4.4.3-4 Return-to-Earth Sequence for Baseline Mars Expedition

Table 4.4.4 -1 Components of the Mission Assurance Program

High-reliability Hardware Parts control Quality Assurance programs Qualification Testing Burn-in of components, subassemblies, black boxes On-orbit Total System Checkout and Runout **Crew Selection and Training** Pre-mission isolation/screening studies of team Space Station Freedom tour of duty Simulations (including re-certs during interplanetary transfers) **Active Redundancy** Triple simultaneous Control Computers (ala STS) Manned back-up of automated systems **Excess Capacity** Propellant sizing for fallback options (fly-around, etc.) 4 parachutes (3 adequate) **Pre-utilization Checkout** Test ignitions of TMIS, MDV, MAV, TEIS, ECCV Mars-critical systems tests prior to MOC/Fly-around decision **Back-up Men and Machines** Crew: Two of each skill; cross-training; on-board learning Hardware: Dual habitats; ECLSS dual- to quad-redundant; Dual MDVs Overlapping missions (including Convoy option)	**On-board Monitoring/Maintenance/Repair** Expert system monitors, diagnoses incipient problems Crew-member specialists, computerized manuals, ground-assisted instructions Tools (general and specialized) Subassembly sparing Stock of standardized parts; Emergency Cannibalization **Alternative Capabilities (Fallback Modes)** LSS consumables oxygen, water, power, heat from LH2/LOX propellant multiple food caches (IMM, MLMM, MAV, RVR) Multiple communication links, transceivers Module and equipment shedding; propellant re-allocations **Crew Compartmentalization** Separated living and working quarters; Isolating doors; Dual egress paths Rotation of crews to surface **Safe Havens** In-space: Nodes; sealed MTMs; MDV; ECCV On-Mars: MAV; RVR Radiation storm shelters for SPE (including explosive trenching on surface) **Abort Modes** TMI abort; MOC fly-around MAV abort during MEL Alternative science if cannot land

4.4.5 Useful Life

No reuse of transportation vehicles is required by the Mars Expedition case study. Reuse of the TMIS engines occurs if the Shuttle-Z concept is utilized. Also, the TEIS engines are used for periapsis raise and orbital maneuvering in addition to the trans-Earth injection burn. All man-rated systems of the MPV must be operational for at least the length of the longest abort mode, which could reach 760 days. Long-lived ECLSS equipment will be a key development for this mission.

4.5 DEVELOPMENT SCHEDULES

The development schedule for the Mars Expedition mission assumes major Phase A starts by 1992 and the first cargo mission launch 2001, followed by a manned launch in 2002. The key development programs, which can be accomplished as one project or as parallel projects, are scheduled in the plan shown in Figure 4.5-1.

4.6 TECHNOLOGY NEEDS

4.6.1 Technical Description

The most pressing technological need is for development of aerobrake technology for the high L/D biconic at Mars. At Mars, the atmospheric entry velocity for aerocapture will be 8.0 km/s for the piloted vehicle. Although the MCV can serve as a pathfinder by achieving aerocapture the previous year, it will be at the significantly lower velocity of 6.9 km/s due to its conjunction class trajectory. High temperature rigid insulation interacting with a

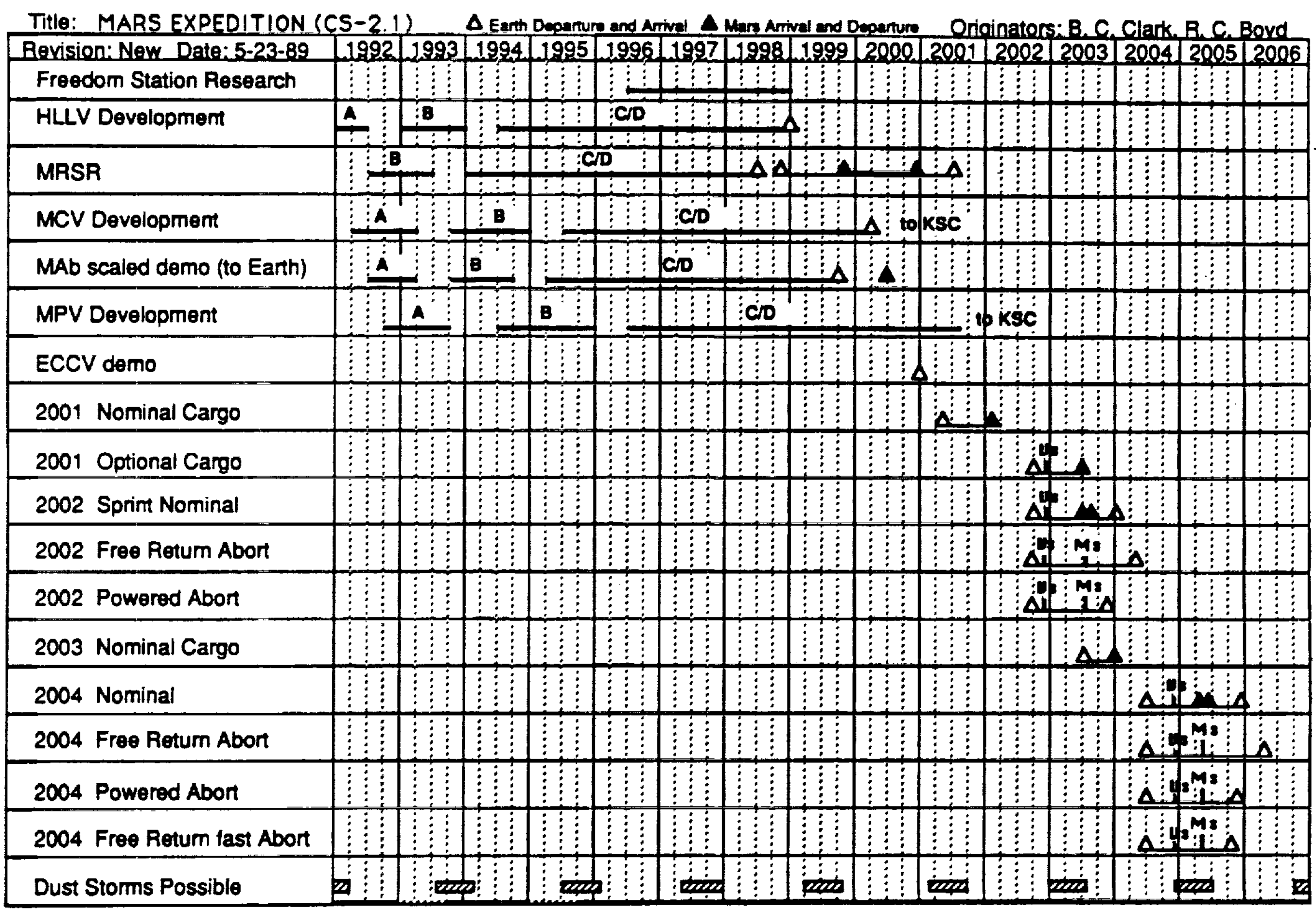


Figure 4.5-1 Mars Expedition (CS-2.1)

CO_2 dust-laden atmosphere may be more complex than the same materials in Earth's atmosphere. Laboratory research and perhaps an aerodynamic test at Mars similar to the Aeroassist Flight Experiment (AFE) aerobrake entry planned for the Earth's upper atmosphere can significantly augment the engineering knowledge presently at hand to increase the confidence with which a minimum mass aerobrake can be designed. An Earth-entry (planetary return) ablative aeroshield will be needed for the ECCV. Cryogenic H/O propulsion and in-orbit handling will also require development, as will long-duration life support systems.

4.6.2 Need Dates

Technology to support aerobrake design should be available by mid-1996 to support MCV and MPV development to achieve launch readiness by the 2001/02 period. This is under the circumstances of a highly success-oriented schedule.

4.7 PRECURSOR NEEDS

4.7.1 Data

Information on candidate landing sites on Mars is needed in order to ensure a high probability of a safe landing. This includes high-resolution imagery, preferably to 0.25 m pixel size and at more than one sun elevation angle, of several alternate sites. Contiguous mapping passes at any one site must encompass 2 x 2 km around the nominal target point, with imaging samples at the 20% to 50% coverage level for the surrounding topography to a distance of at least 25 km. The purpose of this coverage is to certify that landing site relief is well below the 1-2 meter ground clearance of the lander, to determine the geologic setting of the site so as to understand the nature of the surface materials, and to provide for safe alternate touch-down points in the event of an anomalous descent profile.

In addition, it appears to be an absolute requirement that samples of martian soil be aseptically returned to Earth for verification that no toxic or biologically deleterious components are present before allowing man to interact directly with the environment. This sample return mission need not necessarily occur before initiating a Mars project to send humans to Mars, but there must be adequate time to conduct verification tests under varying conditions before committing to a manned launch.

4.7.2 Infrastructure

A heavy lift launch vehicle is a clear requirement for enabling Mars missions because of the impracticality of lifting >500 tonnes in the 20 tonne quanta possible with current United States ETO capabilities. The HLLV must be in-place and operational sufficiently prior to the cargo launch to ensure overall mission success. It is recommended that this capability be achieved by late 1998, and no later than the turn of the century in order to support this mission.

4.7.3 Demonstrations

If the Mars Rover Sample Return (MRSR) mission chooses the same high L/D=1.0 aerobrake, which they are also currently considering, it can provide a demonstration of Mars aerocapture, although at somewhat lower entry velocity than for the MPV. It is also important that this mission test various materials for degradation and interaction with the martian surface environment.

4.8 HUMANS-IN-SPACE RESEARCH NEEDS

To successfully conduct this mission in zero gravity, it is necessary to make very long term investigations of countermeasure effectiveness under these conditions. Because the nominal mission takes almost 16 months, and an abort mode requires over 23 months, the exposure time to weightlessness could be as much as two years. It will be extremely shorter than this time period, assuming they will be conducted on Space Station Freedom, yet there could be reluctance to leave a statistically-significant number of astronauts in LEO for such a long time.

4.9 TRADE RESULTS

Developing very high specific impulse performance for cryopropellant engines has only a minor effect on required propellant mass. For example, if all cryoengines were upgraded to $I_{sp} = 480\ lb_f\text{-}s/lb_m$, the decrease in IMLEO would be only 23.7 t, or less than 3% of total IMLEO. On the other hand, reduction of the TEIS tankage factor from 15% to 7.5% saves 2.5 times this amount. Or, by reduction of TMIS boiloff to a negligible value and moving from high boiloff to low boiloff rates for the TEIS results in an IMLEO savings of more than 3.5 times the amount gained by high performance engines. A synopsis of various boiloff assumptions is contained in Figure 4.9-1. The conclusion is that a number of improvements in efficiency of the overall system can be made in system structural and thermal design, with greater leverage than is to be gained by concentrating on the engines alone.

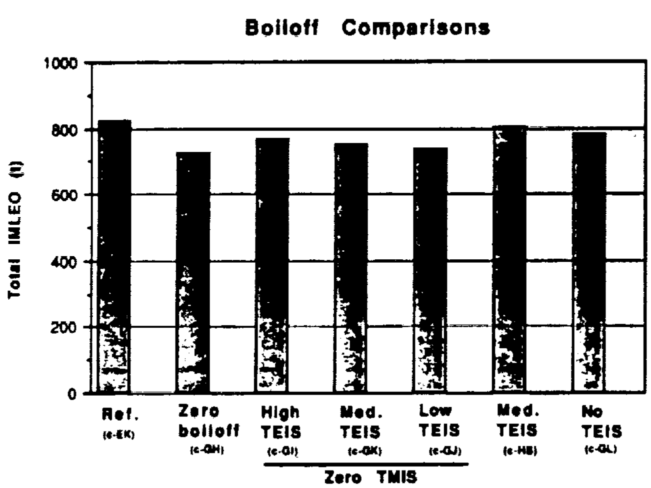


Figure 4.9-1 Boiloff Comparisons

Mars aerobrake mass ratios may be extremely high while maintaining a net gain in IMLEO reduction over all-propulsive capture at Mars, as shown in Figure 4.9-2.

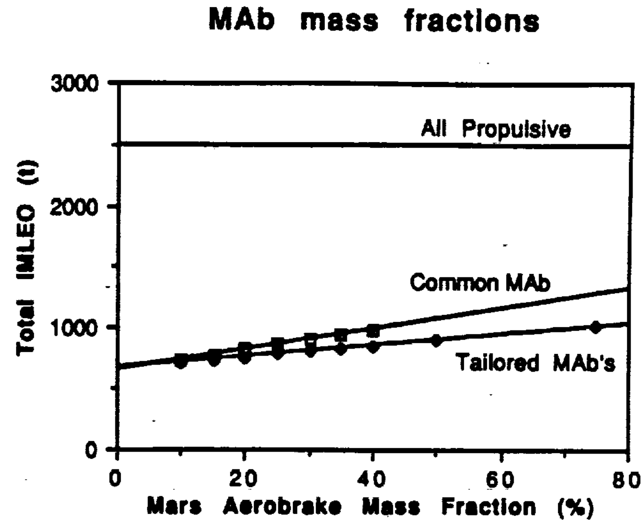


Figure 4.9-2 Aerobrake Mass Ration vs. IMLEO

Providing galactic cosmic ray shielding of 5 g/cm^2 around the entire habitat would increase IMLEO by 109 t, or about 13%. If it were required to add 24.4 g/cm^2 around the habitat, as one researcher has suggested, the penalty would be 485 t of inert mass!

4.10 OPTIONS AND ALTERNATIVES

The ECCV is baselined for direct entry. Unfortunately, because of the high encounter velocities, the deceleration loads will significantly exceed 10-g for the abort mode Earth arrival C_3 of 116 km^2/s^2. If instead of accomplishing a direct entry the ECCV makes a higher pass through the atmosphere and aims for a highly elliptical orbit with 4-day period, the deceleration forces could be as low as 5.5 Earth-g, and less than 10.5-g if the flight path boundary is no lower than 9 km beneath the skipout boundary. Subsequent aeropasses could then be used to circularize the orbit to allow retrieval to SSF.

A number of mission options are examined in Figure 4.10-1. Reduction of the allocated trip time or launching in 2004 have little or no effect on or launching in 2004 have little or no effect on IMLEO. If the MAb is retained and the MPV recovered at Earth ("ETV Recovery"), significant increases in IMLEO are required to provide the TEIS propellant and consequent increased TMIS propellant for retaining the MAb. Note that an artificial gravity system would *not* invoke a large mass penalty on the system.

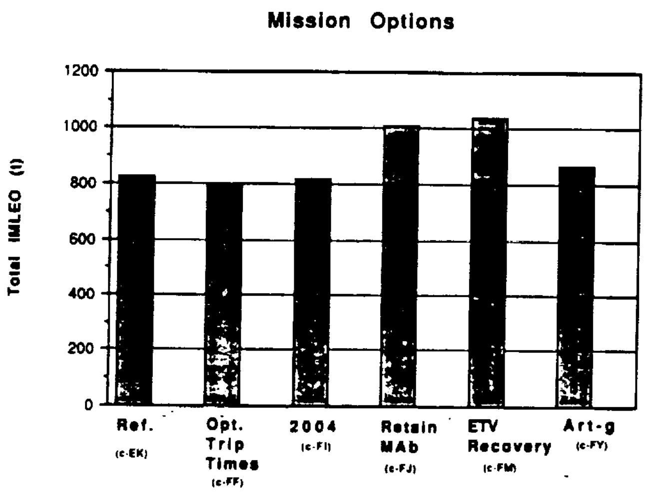


Figure 4.10-1 Mission Options

Several split strategies, a high science payload, a blunt cone low L/D aerocapture brake design, alternative packaging of the MDV in its aerobrake, a tailored brake for the MDV, toroidal propellant tanks, cylindrical habitats, various aerobrake sizes, traditional split mission operations, a two-stage MAV system, and a split crew for landed visits have also been investigated. These options and alternatives are reported in the "June 2nd Drop" document.

4.11 CASE STUDY SUMMARY AND CONCLUSIONS

The high L/D aerobrake can be accommodated if the "fly-out-of-the-brake" approach is followed. Otherwise, a number of confinement problems complicate the engineering implementation. These complications include the thermal control, EVA operations, and solar array management. It is also to be expected that safety issues will arise with the closed-in configuration. However, adopting the strategy of entering the aerobrake just prior to Mars encounter and shedding the brake after the aeropass appears to be a satisfactory approach. Compared to the low L/D aerobrakes, this configuration has the major advantage that on-orbit assembly or deployment is not required. Rather, the dry system can be launched inside the aerobrake, which can also be used as an Earth-to-orbit ascent shroud. For the configuration studied, the ETO vehicle shroud length is slightly exceeded by the aerobrake length.

Alternatives such as the all-up mission, which incorporates the Mars Descent Vehicle into the aerobrake, revealed additional problems in the high L/D approach because of the difficulty of meeting center of mass location constraints for proper stability of the system during aerocapture. Analogously, a number of other factors could result in a heavier and larger vehicle. Any increase in crew size is expected to necessitate more habitat volume. Depending upon the volumetric packing efficiency, the detailed design of the thermal control system for cryopropellant tankage could result in size growth. Accommodating satellites, interplanetary science experiments, or back-up communications antennas would require larger volumes. Thus, this approach does not provide scars for hardware evolution and growth. An upsizing of the aerobrake would be required, and the ETO maximum of 12.5-m diameter would be violated.

Successful jettison of the aerobrake prior to TEIS is critical. The TEIS propellant is inadequate to achieve the required departure asymptote if the aerobrake mass cannot be eliminated. De-docking from the aerobrake will also be critical for thermal, power, and perhaps other engineering systems.

Controlling boiloff of TEIS propellant is an important factor in reduction of IMLEO. Improving specific impulse of TMIS and TEIS engines is also a factor, but not as urgent as advances in the state of the art of boiloff control, which has had relatively lesser impetus for development prior to consideration of manned Mars missions.

Radiation shielding is readily achieved for a solar particle event shelter without significant IMLEO penalty if equipment and consumables are arranged properly. This "pantry-rad shelter" concept can provide 25 g/cm^2 rather than the 5 g/cm^2 value specified. The former value is highly desirable to provide shielding against the rare, but highly hazardous anomalously large flux and energetic solar flare radiation events.

Science IMLEO penalty is negligible. Major sets of science equipment augmented in each mission phase add only about 6% onto IMLEO. The cost of this science is a factor of 10 to 100 times more expensive than launch costs. Therefore, the major limitation on science appears to be the instrumentation and equipment man-rating costs relative to overall program costs.

Several promising alternative mission designs have been identified. The all-up mission is feasible with only about a 5% impact on IMLEO, but aerobrake redesign may be necessary. An Opposition Class mission would save in IMLEO with concomitant increases in total trip time. A conjunction class trajectory mission could be accomplished for less mass with larger crews, more science, artificial gravity, and about an order of magnitude increase in staytime at Mars while lengthening the programmatic time for development of the flight system (by eliminating the precursor cargo launch).

Sprint missions departing Earth in 2002 and 2004 tend to arrive at Mars in the dust storm season. Unless pre-placed surface beacons are provided in the selected area of landing, a descent to the surface may be too hazardous to consider. Conjunction class trajectories are not constrained by dust storm activity because the staytimes are always longer than the dust storm season.

5.0 CASE STUDY COMPARISONS

5.1 COMPARISONS OF RESULTS OBTAINED BEFORE THE MASE SYNTHESIS

It is instructive to compare results obtained by the three case studies described in Sections 2.0 through 4.0. Many specific differences were purposely chosen by MASE in preparation of the SRD to "drive out" the consequences of various types of requirements. However, because of the multiplicity of differences between scenarios, it is not possible to reach firm, generalized conclusions regarding any individual issue that is approached differently in two scenarios. For example, IMLEO is driven by a host of factors and it is not appropriate to conclude that conjunction class trajectories require higher mass than sprint class trajectories, just because some CS 5.0 missions give higher total IMLEO than the CS 2.1 mission. With these caveats in mind, the following comparisons can be made.

Lunar vehicles are lighter in weight because of their very small crew cabs compared to the much larger habitats of the interplanetary cruise Mars vehicles. For example, the heaviest Lunar vehicle (dry weight) is only slightly over 15.5 t, whereas the Mars piloted vehicle for CS 2.1 is 81 t. For CS 5.0 (Mars Evolution), the dry MPV (without cargo) is 96 t.

Propellant capacities also differ, but not as significantly. The lunar vehicles each hold 59 t of H/O propellant. The MPV of the Mars Expedition holds 96 t of propellant, while the analogous vehicle of the Mars Evolution holds 79 t. However, it must be remembered that these quantities are for TEIS and orbital ops for the Mars vehicles, but do not include the TMIS propulsion requirement. In the case of the Lunar vehicles, the TLIS tanks are combined with the TEIS, and no staging is involved. Therefore, the CS 4.1 loads include both outbound and inbound propellants. TMIS propellants for the CS 5.0 Mars Evolution vehicles vary, ranging from 235 t to 395 t. For the Mars Expedition, TMIS propellant is 381 t for the MPV and 119 t for the MCV. It is mainly because of these large propellant loads that Mars missions will need a new, high-capacity HLLV with 100 to 150 t Earth-to-orbit payload capability, whereas a Shuttle-C or other class carrier with ~68 t capacity is easily adequate for Lunar missions.

Lander vehicles are also quite different for the three case studies. Whereas Lunar landers have dry masses less than 3.5 t (except for the propellant tanker, which is still <5.0 t), the Mars landers are 18.5 t for the Expedition case's MDV and about 15 t for the Evolution case's MCSV. These comparisons are not valid without many qualifications, however, because of the fact that no cargo is carried in the 3.5 t mass of the LCL (which has a 20 t down capability). As has been previously shown, and is reinforced in the present studies, Mars and Lunar landings are asymmetrical in requirements. Because of aeroassists in landing, the Mars landers require less propellant whereas the 2000 m/s ΔV for lunar landing invokes more propulsion and also places much greater requirements on the throttling range of the descent engines. For the lunar case, this range exceeds 20:1 for many engine cluster and failure mode scenarios. For the Mars cases, the range is nominally 3:1.

Operations are quite different for the three studies as performed. However, this was much more a function of the groundrules assigned to each case study rather than resulting from transportation's derived requirements: Lunar transfer vehicles were baselined as reusable and stored/serviced at Space Station Freedom (SSF); Mars Expedition vehicles were assembled and fueled with no node support; Mars Evolution vehicles were re-usable, but with a special node created independently of SSF. The lessons learned from this exercise include the fact that Freedom Station might undergo significant impact if a major role to support manned exploration missions becomes a requirement. Another clear result is that because all three transportation systems are highly dependent on H/O cryopropellant (specified by SRD) and because the loads are large compared to the HLLV capability, some form of on-orbit propellant transfer is required. Whether a zero-g acquisition system or acceleration-forced transfer is employed is more or less independent of the mission objective. Low boiloff is a major objective for the Mars mission's TEIS, but not nearly so much a concern for the the TMIS, TLIS, and lunar return TEIS because of the shorter time durations involved.

Communications are significantly different for the lunar compared to the Mars missions, because of use of the Deep Space Net for the latter. Mission operations are also quite different because of up to 40 minutes of roundtrip delay time for Mars compared to 3 seconds for the moon. Mars missions will require a greater autonomy and authority of the crew with respect to controllers on Earth, whereas the lunar missions can be mainly directed by ground personnel. Again, because of the distances and time isolation, rescue strategies are fundamentally different for the two approaches. A Lunar rescue vehicle could be staged on the moon, in LLO, or at SSF. In any event, even without such a vehicle, if a LCV were available and a spare crew cab were maintained at Freedom it would be technically possible to outfit a rescue flight on an effective time scale (days). Rescue flights to retrieve Mars astronauts are out of the question except on time scales of months to years.

5.2 MASE SYNTHESIS OPTIONS AND COMPARISONS

Post-dating the completion of Cycle 2 studies, the MASE team synthesized options to each Case Study. The following summarizes the additional studies provided by the Transportation Integration Agent in support of this activity.

5.2.1 Lunar Evolution

During the MASE synthesis activities it became evident that commonality between the cargo and piloted vehicles could be accomplished not just at the subsystem level, but at the integrated vehicle level as well. Vehicle commonality was desired to reduce the number of different vehicles and to simplify the refurbishment and servicing operations by employing similar vehicles designs. The common vehicle design approach utilizes the cargo vehicles as the propulsion stage for the piloted vehicle. The only difference between the cargo and piloted vehicles was the addition of a "bolt-on" crew module for the piloted mode of operation. This simple design philosophy not only reduces the number of vehicles required but at the same time increases the payload delivery capabilities of the piloted vehicles. In order to differentiate the common vehicles from the vehicles described in section 2.0, the synthesized vehicles are termed the Lunar Transfer Vehicle (LTV), for transportation between the Earth and Moon, and Lunar Excursion Vehicle (LEV), for transportation between lunar orbit and the lunar surface.

Lunar Excursion Vehicle Design—Emplacement of the initial lunar outpost requires numerous flights of the cargo Lunar Excursion Vehicle (LEV-C) which was designed to carry 20 t of cargo in the low-lunar orbit (LLO) fueled mode. It was desired to reduce the number of up-front flights by improving the payload capability of the vehicles during the initial flights. Therefore, the LEV-C was redesigned to carry the 20 t of cargo when utilizing lunar derived oxygen. The vehicles are also expended during the early flights in order to further increase the delivery capabilities and to serve as testbeds to better understand what types of servicing will be required and how this servicing will be done. The payload delivery capabilities of the vehicles are shown in Table 5.2.1-1. The propellant tanks for the LEV were increased in size to accommodate the new payload delivery requirements. In addition,

Table 5.2.1-1 Scaled-up Lunar Evolution Vehicles Loadings

Mission	Vehicles Used	Operational Mode		Vehicle Dry Mass		Propellants Req. *		Payload Carried	
		LTV	LEV	LTV	LEV	LTV	LEV	LTV	LEV
Initial Cargo Mission	1 LTV-C 1 LEV-C	Expend in LLO	Expend in LS	7.8 t	3.4 t	Earth: 124.1 t	Earth: 24.2 t	64.6 t	37.0 t
Consolodiation Phase Cargo Mission	1 LTV-C 1 LEV-C	LEO Based	LS Based	9.7 t	3.4 t	Earth: 125.8 t	Earth: 24.9 t	57.9 t	33.0 t
Use Phase Cargo Mission Using LLOX in LEVs	1 LTV-C 2 LEV-C	LEO Based	LS Based	9.7 t	Per LEV-C 3.4 t	Earth: 106.8 t	LEV-C: Moon: 20.4 t** Earth: 3.4 t***	46.8 t	Per LEV-C 20.0 t
Initial Personnel Mission	1 LTV-P 1 LEV-P	Expend in LEO	Expend in LLO	18.7 t	6.4 t	Earth: 146.6 t	Earth: 24.8 t	54.7 t	23.5 t
Consolidation Phase Personnel Mission	1 LTV-P 1 LEV-P	LEO Based	LS Based	18.7 t	6.4 t	Earth: 135.7 t	Earth: 24.8 t	48.5 t	23.7 t
Use Phase Personnel Mission Using LLOX in LEVs	1 LTV-P 1 LEV-P 1 LEV-C	LEO Based	LS Based	18.7 t	LEV-P 6.4 t LEV-C 3.4 t	Earth: 124.8 t	LEV-P Moon: 21.2 t** Earth: 3.5 t*** LEV-C Moon: 20.4 t** Earth: 3.4 t***	42.1 t	LEV-P 15.2 t LEV-C 20.0 t

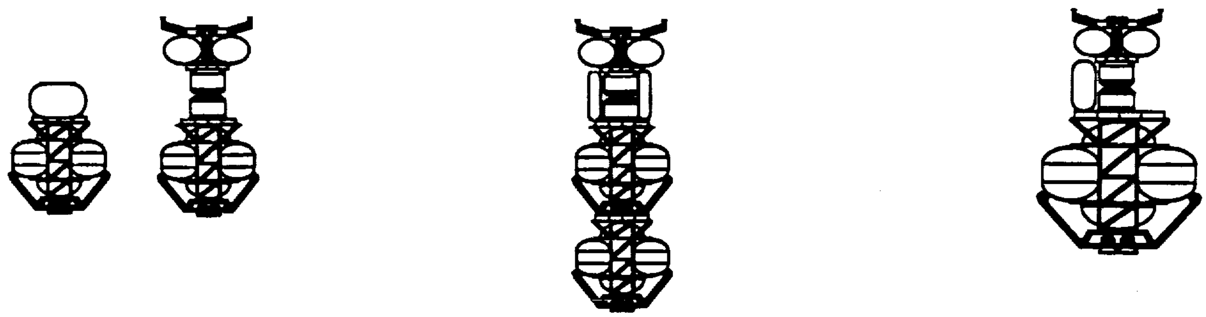
Notes:
* Source: Amount
** LLOX
*** LH2

the landing structure was modified allowing it to handle the increased landed mass. Since lunar derived oxygen was not used in the LTVs, the tanker version of the LEV was not required, and therefore was not redesigned.

Lunar Transfer Vehicle—Another important trade conducted during this case study addressed the sizing of the LTVs. The alternatives studied were sizing the LTVs for the utilization phase when LEVs are using lunar liquid oxygen (LLOX), versus sizing the LTV for the initial flights of the emplacement phase. Figure 5.2.1-1 illustrates the different options and the issues associated with each. Most importantly, it was desired to reduce complex operations and vehicle servicing requirements during the initial emplacement phase of the case study.

VEHICLE STACK AT TRANS-LUNAR INJECTION

Caravan	Staging	Scaled Up



Concerns During Initial Flights

Caravan	Staging	Scaled Up
Complicated Initial Operations	Simplified Initial Operations	Simpliest Initial Operations
2:1 OTV Payload Ratio	2:1 OTV Payload Ratio	1:1 OTV Payload Ratio
Vehicle Processing-Doubled	Vehicle Processing-Doubled	Simplified Vehicle Processing
Autonoumous Rendeveous	No Autonoumous Rendeveous	No Autonoumous Rendeveous
Autonomous Payload Transfer	No Autonomous Payload Transfer	No Autonomous Payload Transfer

Concerns During Use Phase

Caravan	Staging	Scaled Up
Simplified Use Phase	Simplified Use Phase	Optional Use Phase
1:1 OTV/Payload Ratio	1:1 OTV/Payload Ratio	1:1 or 1:2 OTV/Payload Ratio

Figure 5.2.1-1 Lunar Transfer Vehicle Sizing Options

For the caravan and staging phase options, the LTVs were sized to deliver to LLO one LEV payload and enough hydrogen for one LEV mission. Although this projection gives a favorable 1:1 LTV to LEV in LLO ratio per mission for the utilization phase, the limited payload mass capabilities of the LTVs dictates that two LTVs be used to deliver a fully loaded and fueled LEV during the initial flights of the emplacement phase. Also, during the remainder of the emplacement phase and throughout the consolidation phase, two LTVs are still required to deliver one LEV payload and enough LEV propellant (oxygen and hydrogen) for one LEV mission. Thus, the majority of the missions during this case study would have a 2:1 LTV to LEV in LLO ratio per mission. As shown in Figure 5.2.1-1, these missions can be accomplished in either a caravan or staging option. The basic problem with the small LTV in either the caravan or staging option is that it places the most difficult operational demands on the Earth-Moon transportation infrastructure at the beginning of the outpost's development, when little or no experience exists. Complex autonomous rendezvous, docking, and payload transfer operations must occur on the first lunar mission with the caravan option, while simultaneous processing of two LTVs per lunar mission is necessary for both caravan and staging options. Thus, the small LTV sizing option greatly impacts the activity level and size of the servicing facility at Space Station Freedom at the very beginning of the case study, since two LTVs need to be concurrently processed for each mission to LLO.

To decrease the number of LTV flights, and thus decrease the associated LTV activity at Freedom, a larger LTV was considered that would always allow for a 1:1 LTV to LEV ratio per mission. The larger LTV would also allow for a gradual increase in capability at Freedom, where the initial missions to the Moon could be accommodated with a less complex, single LTV facility. The scaled-up LTV was sized to deliver a fully loaded and fueled LEV from LEO to LLO. Once LEVs are based on the lunar surface, one LTV delivers one LEV payload and enough LEV propellant (oxygen and hydrogen) for one LEV mission.

During the later portions of the utilization phase when the LEVs are using LLOX, the larger LTV has the ability to deliver two LEV payloads and enough hydrogen for two LEV missions, or it can continue to deliver enough payload and hydrogen for one LEV. Thus, the LTV to LEV ratio could be 1:2 or 1:1 for this portion of the case study. Because of the less complicated LEO operations associated with a 1:1 LTV to LEV ratio and the fact that fewer larger LTVs are needed for the case study than smaller LTVs, a scaled-up Lunar Transfer Vehicle was designed so that it could accommodate an all-up delivery of this payload, including the LEV and its propellant load. The resulting vehicle is shown in Figure 5.2.1-2 (piloted version). Payload delivery capabilities of this new system are again given in Table 5.2.1-1.

Several options were available to configure the lunar transfer vehicle (LTV). Note that the foldable aerobrake strategy is preserved. The number of cryoengines is increased from two to four, which is also the number of engines used on the landers. Instead of keeping the internal truss with multiple tanks, which has the advantage of allowing on-orbit changeout of a single tank, only two tanks are supplied, one each for the oxidizer and fuel. This has the advantage of easing the engineering difficulty of achieving acceptable thermal control of the cryogenic liquids by providing a minimal area-to-mass ratio and hence reducing the heat load on the tanks. In addition, the fully-enclosing support skirt provides the dual use of not only structural support but also meteoroid shielding. On return to Earth, temporary supplemental shielding for orbital debris protection can be added around the skirt, analogous to hanging a curtain. Note that the tank diameters are constrained so that the folded configuration remains compatible with the 10-m diameter launch shroud of the ETO vehicle. A drawback of this approach is the high center of gravity (c.g.) on aerocapture back at Earth. However, the new c.g. location remains compatible with the requirements for control stability with respect to location of the center of pressure. This is made possible by the high L/D for the aerobrake shape selected.

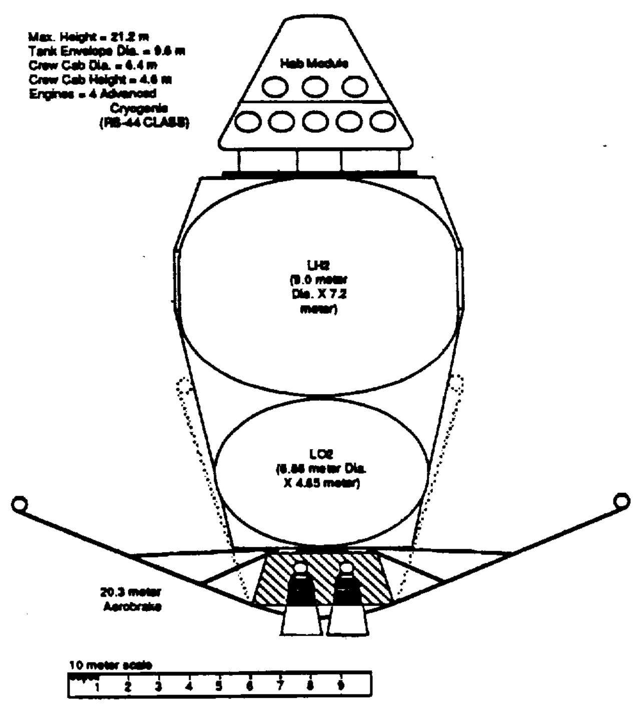


Figure 5.2.1-2 Upsized Lunar Transfer Vehicle for MASE Synthesis Option

5.2.2. Mars Evolution

This case study was significantly recast in mission sequence, payload delivery requirements, and the choice of propulsion technologies utilized. These substantial changes to the case study architecture had a major impact on the transportation vehicles' design and the way that they were utilized. The details of the reference integrated mission are presented in Volume I, Section 3.2.

The interplanetary Mars Transfer Vehicle (MTV) was kept the same (both Piloted and Cargo versions), as was the Mars Cargo Lander (MCL). The MCL is expendable and designed to deliver 50 metric tons of cargo to the Mars surface. The vehicle used for personnel transport to and from the Mars surface was significantly redesigned. First, it was changed from a fully reusable to a two-stage, expendable vehicle. This change was necessitated by the new requirement for this vehicle to deliver 25 metric tons of cargo to the surface in addition to the crew. The same vehicle was also designed to transport the crew between the Mars surface and a high elliptical orbit or a Phobos-compatible orbit. The Mars Crew Sortie Vehicle and its constituent vehicles were thus eliminated, as was the Phobos/Deimos Excursion Vehicle. Also, the tether-assisted momentum transfer at Phobos for descent and trans-Earth injection was not implemented in the MASE synthesis option. Finally, the NTR and NEP options for piloted and cargo vehicles were not implemented in the MASE integrated mission.

The first flight to Mars in the MASE integrated mission launches in 2005 and is an unmanned cargo mission which delivers the Mars surface habitat module and associated equipment. A total surface payload of 43.4 metric tons is delivered to Mars on the MCL. Flight 2 is an all-up opposition class mission launching in 2007, with a crew of four all of whom descend to the surface for a 30-day stay. The crew is transported between the elliptical parking orbit and Mars surface in the redesigned Piloted Mars Excursion Vehicle. This vehicle uses LOX/LH2 in the descent stage and storable bipropellant in the ascent stage. The crew lives in the lander in a separate habitation module specifically designed to support them for their thirty day stay while they emplace the permanent habitat module delivered on Flight 1. This habitation facility is carried on the MEV-P as payload and is not an integral part of the vehicle design since it is used only this one time on Flight 2.

Flight 3 is a piloted mission with the a crew of four spending over 500 days on the surface of Mars. The MEV-P delivers 24 metric tons of cargo in addition to the crew to the Mars surface on this flight. Flight 4 is an unmanned cargo mission that delivers the propellant plant and associated equipment to Phobos (55 metric tons total) and also delivers an additional 41.3 metric tons of cargo to the Mars surface on an MCL. Piloted missions with extended stay times are undertaken on Flights 5 and 6. On Flight 5, the crew size is increased to 5, and the MPV rendezvous with Phobos so that the crew can perform their primary mission function of emplacing the Phobos propellant plant and beginning its production. The crew then performs a global reconnaissance of Phobos before departing to the Mars surface in the MEV-P. On Flight 6, the MPV also rendezvous at Phobos so that the Phobos-supplied TEI propellant can be loaded onto the MPV. The crew then descends to the Mars surface in the MEV-P for a 500-day stay.

Flight 7 is an unmanned cargo flight that delivers over 100 tons, including the constructible habitat and surface ISRU facilities, to the Mars surface using two MCL's. Flight 8 is the final mission manifested in this year's case study and it assumes crew size growth to seven and a 500 day stay on the Mars surface.

The detailed manifests of these missions are provided in Volume I, Section 3.2. The results are summarized in Table 5.2.2-1, and provided in more detail in Appendix F. Two discrepancies exist between these data and the MASE result in Volume I. First, the Flight 1 IMLEO is approximately 25 metric tons greater in the MASE version due to a late post-synthesis increase in the TMI delta-V that was not accommodated in the TIA results. Second, the Flight 4 IMLEO is more than 70 metric tons greater in the MASE version because the 50 metric tons of payload delivered to the Mars surface (in addition to the cargo delivered to Phobos) were not originally included in the TIA calculations.

Table 5.2.2-1. Mass Allocation Results for Mars Evolution MASE Option

Launch Date	Mission Type	Mars Orbit	IMLEO (tonnes)
8/2004	cargo	500 km circ	522.1
9/2007	piloted, Op	250 x 1 sol	1051.8
10/2009	piloted, Cn	250 x 1 sol	798.9
11/2010	cargo	Phobos	573.3
12/2011	piloted	Phobos	1045.2
1/2014	piloted	Phobos	721.4
2/2016	cargo	Phobos	685.3
3/2016	piloted	Phobos	739.0

5.2.3 Mars Expedition

The major difference for the Mars Expedition is the decision to investigate as synthesis option the approach of an all-up sprint class mission, i.e., no separate launches of any cargo. A configuration for this approach is shown in Figure 5.2.3-1. Recalculations of mission masses for MASE specified conditions are shown in Table 5.2.3-1 and Appendix F. From these, the TEIS tanks were resized. It was found that the length of the vehicle increases by less than 1.5 m, which is compatible with maintaining the general biconic shape, the specified L/D of 1.0, and the vehicle and aerobrake maximum diameter to 12.5 m. However, a more detailed analysis would be necessary to determine whether the center of mass is within acceptable fore-aft bounds for the highly constrained requirements of this design.

Table 5.2.3-1. Results for MASE Synthesis Option for Mars Expedition

	500 km x 500 km	250 km x 1 sol
ΔV allocations (m/s)		
TMI	4400	4400
TEI	3900	3450
MDV	700	600
MAV	4200	5800
IMLEO (tonnes)	775.9	762.4

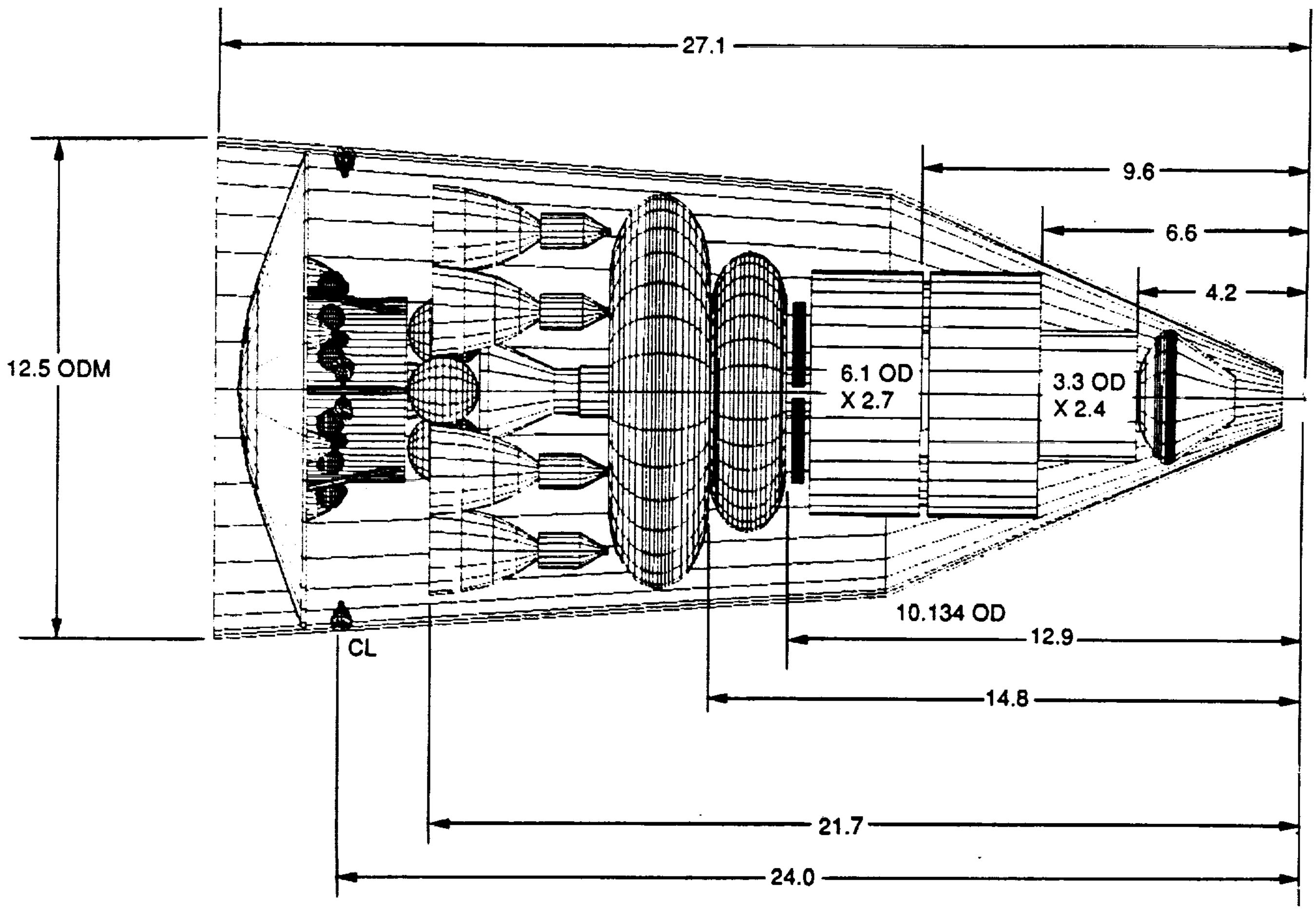


Figure 5.2.3-1 Configuration for Mars Expedition MASE Option

C-2

6.0 CONCLUSIONS AND RECOMMENDATIONS

In the above detailed sections on Case Studies, a number of assumptions, trades, and trends may be found. In this section of this report, the higher level conclusions are discussed, particularly with regard to the larger context of the relative merits and implementations for each case study.

In the Lunar Evolution case study (CS 4.1), it was found possible to achieve a high degree of commonality between transportation vehicles. For example, all vehicles consist of a common central truss, to which are mounted two fuel (LH2) and two oxidizer (LOX) tanks. The basic propulsive system serves for both the piloted and cargo versions of the appropriate vehicle. Space transfer vehicles (STV), shuttling from LEO to LLO and back, use the same propulsion system whether manned or unmanned. Modularity provides the possibility of a detailed design that can accommodate the objective of preserving commonality and simply adding a cargo pallet or a crew cab. A similar approach obtains for the lander vehicles. The same type of advanced space engine is baselined for each, although four engines are provided for landers while only two engines are needed for the transfer vehicle. An improved, but near-term advanced engine is appropriate. Key operating parameters of this engine include long life with multiple restart capability, wide throttling range, and sufficiently high chamber pressure to achieve high specific impulse yet provide a small volumetric envelope. The case study values are I_{sp} = 4.71 kN-s/kg, T = 66.7 kN within size constraints of 1.0-m diameter by 1.6-m long.

It is recommended that rescue capability be provided for the Lunar Evolution case. A straightforward approach is to always maintain two working vehicles in LEO (or LLO or Lunar surface). If a spare crew cab is also maintained, it can be switched out with the payload on the cargo vehicle to convert it into a rescue vehicle or to simply augment the manned transportation capacity.

It is recommended that a one-engine out criterion be adopted for cryoengines. If a dual fault tolerant criterion is to be maintained, it will have major impacts on system design, and could also drive the aerobrake to a higher weight in order to certify the brake as non-degradable structure.

These studies, supplemented by independent analyses, indicate that lunar liquid oxygen (LLOX), may not payoff for export to Earth by chemical propulsion techniques. However, LLOX may produce a net reduction in cost if employed for lunar-based use, such as ascent, descent, and return to Earth. These studies also indicate that the leverage obtained by increasing the oxidizer/fuel ratio from 6:1 to high values such as 8 or 10:1 is not large and does *not* indicate a need for a high O/F ratio engine.

For both the Lunar and Mars Evolution studies it was possible to package the vehicles inside a 10-m diameter HLLV payload shroud, with subsequent automated deployment (in the case of Mars Evolution) on-orbit. Because of the specified high L/D aerobrake for the Mars Expedition case, and the other requirements on crew size and payloads, it was not found to be possible to achieve this low value; rather, a 12.5-m diameter shroud appears to be the minimum acceptable size. It also can be concluded that evolution and growth are not readily achieved with the Mars Expedition case unless the 12.5-m diameter requirement is waived. Even with the size selected, it was found to be difficult to package an interplanetary vehicle for a crew of 3 together with return TEIS within this maximum diameter limitation.

One of the most significant conclusions from the Mars Expedition case study is that it does appear possible to accommodate a modest mission within the specified high L/D aerobrake constraint. To eliminate significant problems with thermal control, safety issues, asymmetric solar panels (necessitating counter-torquing with ACS), and propellant transfer it is important to include the capability for de-docking and re-docking of the main vehicle with its aerobrake. It is recommended that the design be such that the brake need only be entered just before the aerocapture event. The mission is benefited if the brake can be discarded immediately after the aeropass, and it is quite critical that the brake be jettisonable before the TEIS burn. Otherwise, the propellant capacity of the TEIS must be increased significantly, resulting in an IMLEO growth of more than 180 t.

For both Mars case studies, propulsion engine specific impulse is important, but not to a degree as to be a major issue. Cryopropellant storage with minimum boiloff is important if IMLEO is to be kept low. Because an active control system would cause a significant decrease in mission success probability due to the lower reliability of a spaceborne cryo-refrigerator and liquifacation system, it is recommended that only passive control means be considered. This in turn requires that highly engineered passive storage systems be developed to reduce both the tank mass and the rate of loss of propellant.

It was found during the Expedition study that an all-up single mission did not create a large increase in IMLEO, mainly because it had already been baselined that the TEIS would fly with the piloted mission. Increasing the science payload also did not increase IMLEO significantly. The difference between no science and a very major science complement (14 t) does not increase IMLEO by more than 6%.

Radiation shielding for a solar flare storm shelter was not found to be a significant impact if the shielding is confined to a small volume (pantry) and if on-board supplies and equipment were used rather than adding tailored deadweight shielding. It is recommended that at least 25 g/cm^2 of wall thickness be provided since this is well within capabilities and will greatly increase the effectiveness compared to the 5 g/cm^2 specified by the SRD. Conversely it was found, however, that providing 25 g/cm^2 around the entire habitats to reduce the theoretical dose caused by high energy galactic cosmic rays would produce a very major increase in mission mass — at least 500 t.

It should be pointed out that for the mission launch years of 2002 and 2004, piloted sprint missions will arrive at Mars during the season when dust storms are possible. Landings may be more hazardous because of the difficulty in visually acquiring the landing site, either from orbit or during the early part of the descent. If navigation aids in the form of radio transducer beacons can be pre-landed at the site, landings may become possible with acceptable success probabilities.

A near-term advanced cryogenic engine with a compact profile is needed for the Lunar and Mars Evolution crew sortie vehicles (piloted landers) to allow efficient packaging. For the lunar case, extend/retract nozzles are necessary to allow closing of aerobrake doors. The Mars Expedition uses expendable landing propulsion and does not generate this advanced engine requirement.

In the Mars Evolution, the use of the Gateway (Phobos) significantly reduces the IMLEO for the later missions, but causes significant increases in up-front IMLEO to deploy infrastructure at Phobos. The use of Phobos propellant is beneficial, but not as a major IMLEO multiplier except over a very long term and after a large number of missions. The use of tether-assisted momentum transfer at Phobos is also beneficial in reducing IMLEO. In principle, the tether system pays back its own mass after only two mission uses. However, because Phobos propellant is baselined as being available, the addition of a tether system seems unnecessary. The two approaches are duplicative in the final effect of reducing the IMLEO for the missions.

Robotic deployment of the Gateway propellant production plant would not seem feasible at Mars because of the long communication delays, which greatly restrict the rate of operations under Earth-directed control by teleoperation. Rather, astronauts must be at Phobos or in the vicinity to permit direction of the robotic assembly of the plant.

Development of two different types of nuclear propulsion is not synergistic. Once NTR is made available, it can be used in two ways: to reduce the flight time for piloted missions and to increase the throwweight (per unit IMLEO) for cargo missions. The development of NEP is overlapping of the latter capability and does not seem necessary. Because the NTR can be used advantageously for both cargo and manned missions, it is concluded that it would be preferable to NEP for this scenario. The electric propulsion very significantly increases the flight times for piloted transportation vehicles compared to chemical or thermal rocket propulsion approaches.

The reference mission sequence of the Mars Evolution case study is not compatible with the ETO and vehicle technology constraints. Hence, it is recommended that an altered sequence be considered. The 2004 mission has high encounter energetics, which drives the aerobrake design. Yet this occurs only once in all the missions. It is suggested, therefore, that to minimize technology and mass drivers on aerobrakes that the opposition class mission be eliminated or baselined differently.

APPENDIX A. ACRONYMS AND ABBREVIATIONS

Acronym	Definition	Comments
Ab	Aerobrake	
AC or A/C	Aerocapture	
ACS	Attitude Control System	
AFE	Aeroassist Flight Experiment	(aerocapture brake test program)
AI	Artificial Intelligence	
ALS	Advanced Launch System	(HLLV)
ALSPE	Anomalously Large Solar Particle Event	(maximum solar flare)
AOTPM	Ascent and Orbit Transfer Propulsion Module	
APM	Ascent Propellant Module	
ARD	Ascent, Rendezvous, and Docking	
ASE	Airborne Support Equipment	
CCM	Crew Cab Module	
c.m.	Center of Mass	
Cn	Conjunction	(conjunction class trajectory)
DeEV	Deimos Excursion Vehicle	
DMS	Data Management System	
DSM	Deep Space Maneuver	(broken-plane or other major interplanetary propulsive maneuver)
EAb	Earth Aerobrake	(for aerocapture of ETV)
ECCV	Earth Crew Capture Vehicle	(small vehicle for crew EOC)
ECLSS	Environmental Control and Life Support System	
EP	Electric Propulsion	(see also NEP, SEP)
EELS	Earth Entry & Landing System	(can include propulsion and aerobrake/heatshield)
EPS	Electrical Power System	
ETO	Earth-to-Orbit	(vehicles such as STS, HLLV, etc.)
ETV	Earth Transfer Vehicle	(MSS configuration during flight to Earth)
EVA	Extra-vehicular Activity	(any human activity outside protective shirtsleeve environment and requiring a spacesuit)
FRCI	Fiberous Refractory Composite Insulation/ Flexible Resuable Surface Insulation	
FTS	Flight Telerobotic Services	(teleoperated robot for SSF)
GCR	Galactic Cosmic Rays	(cosmic rays, from outside the solar system)
HAL	Hyperbaric Airlock	

HLLV	Heavy-Lift Launch Vehicle	(SDVs and other advanced launchers)
HMF	Health Maintenance Facility	(diagnosis and treatment of illness and trauma)
HMO	High Mars Orbit	(often used to refer to 1-sol elliptical orbit)
IMLEO	Initial Mass in Low Earth Orbit	(mass at first ignition for LEO escape)
IMM	Interplanetary Mission Modules	(Hab/Lab/Log modules for crew in space)
ISE	Interplanetary Science Equipment	(Astrophysics, biological, planetary science equip)
I_{sp}	Specific impulse	(units of kN-s/kg [km/s] or lb_f-s/lb_m)
ISRU	*in situ* Resources Utilization	
IVA	Intra-vehicular Activity	(human activity inside the habitat pressure vessel)
kW	kilowatt	
L1	Lunar libration node 1	
LCL	Lunar Cargo Lander	(cargo carrier, LLO to LSurf)
LCV	Lunar Cargo Vehicle	(cargo carrier, one-way LEO to LLO)
LEO	Low Earth Orbit	(typically 400 to 500 km circular)
LF	Lunar Freighter	(cargo and propellant carrier, shuttling between LLO and LEO)
LLO	Low Lunar Orbit	(typically 100 to 300 km circular)
LLOX	Lunar LOX	(liquid oxygen produced on the moon)
LMO	Low Mars Orbit	(typically 300 to 500 km circular)
LOX	liquid oxygen	
LPL	Lunar Piloted Lander	
LPT	Lunar Propellant Tanker	(propellant transporter, shuttling between LSurf and LLO)
LPV	Lunar Piloted Vehicle	(crew carrier, shuttling between LEO and LLO)
LSS	Life Support System	
LSurf	Lunar Surface	(surface of the moon)
M	mega	(one million)
m	meter	
MAb	Mars Aerobrake	(for aerocapture of MTV)
MAV	Mars Ascent Vehicle	(one-way ascent from MSurf to orbit)
MCC	Mid-course correction	
MCL	Mars Cargo Lander	(cargo carrier, Mars orbit to MSurf)
MCSV	Mars Crew Sortie Vehicle	(crew carrier, shuttling from Mars orbit to MSurf)
MCV	Mars Cargo Vehicle	(unmanned cargo transporter for LEO to Mars orbit)
MDV	Mars Descent Vehicle	(the vehicle which de-orbits to land on Mars)
MELS	Mars Entry & Landing System	(de-orbit propulsion + aerobrake + parachute +terminal propulsion + G&C)

MLI	Multilayer Insulation	
LAPM	Lander/Aerobraking Propulsion Module	
LEO	Low Earth Orbit	
LMO	Low Mars Orbit	
MCL	Mars Cargo Lander	
MCSV	Mars Crew Sortie Vehicle	
MCV	Mars Cargo Vehicle	
MDV	Mars Descent Vehicle	
MLI	Multilayer Insulation	
MLSE	Mars Landed Science Equipment	
MMU	Manned Maneuvering Unit	
MO	Mars Observer	(polar orbiter mission to Mars, planned for 1992 launch)
MOCS	Mars Orbital Capture System	(includes propulsion, aerobrake, GN&C for orbital capture)
MOO	Mars Orbital Operations	(maneuvers for orbit maintenance and orbit alterations)
MOO1	" " "	(MOO for initial maneuvers; e.g., to rendezvous with TEIS or Phobos)
MOO2	" " "	(MOO for final maneuvers; e.g., to prepare for TEI)
MOSE	Mars Orbital Science Equipment	(Instruments for studies from Mars orbit)
MOV	Mars Orbiting Vehicle	(MSS configuration in Mars orbit)
MPV	Mars Piloted Vehicle	(crew carrier from LEO to Mars orbit)
MRSR	Mars Rover Sample Return	(combined rover and sample return mission)
MSurf	Mars Surface	(any location on the surface of Mars)
MTV	Mars Transfer Vehicle	(MSS configuration during flight to Mars)
NEP	Nuclear Electric Propulsion	(ion drive; via nuclear reactor)
NEPF	Nuclear Electric Propulsion Freighter	
NERVA	Nuclear Engine for Rocket Vehicle Application	
NSO	Nucelar Safe Orbit	
NTR	Nuclear Thermal Rocket	
OMV	Orbital Maneuvering Vehicle	
OOA	On-orbit Assembly	
OpVs	Opposition/Venus Swingby	(Opposition class trajectory with one or more swingby of Venus)
ORU	Orbital Replaceable Unit	
P/L	payload	(means different things to different people)

PhEV	Phobos Excursion Vehicle	
PhLOX	Phobos LOX	(liquid oxygen produced on Phobos)
PRFE	Planetary Return Flight Experiment	(hypervelocity aerobrake test; compare with AFE)
prox ops	proximity operations	
PVPA	Photovoltaic Power Array	(solar cell electrical power source)
RCS	Reaction Control System	
RMS	Remote Manipulator System	(Shuttle robot arm)
RTG	Radioisotope Thermoelectric Generator	
RVR	Rover	
SDV	Shuttle Derived Vehicle	(ETO booster whose technology is derived from Shuttle systems)
SEP	Solar Electric Propulsion	
SSF	Space Station Freedom	
SSME	Space Shuttle Main Engine	
STIA	Space Transfer Integration Agent	(see also TIA)
STS	Space Transportation System	(Shuttle)
STV	Space Transfer Vehicle	
TABI	Tailorable Advanced Blanket Insulation	
t	tonne	(metric ton; 1000 kg; 2204 lb_m)
TBD	To Be Determined	
TCS	Thermal Control System	
TEI	Trans-Earth Injection	(Mars orbital escape and trans-Earth maneuver)
TEIS	Trans-Earth Injection System	(propulsion and guidance system for TEI)
TIA	Transportation Integration Agent	(see also STIA)
TMI	Trans-Mars Injection	(Earth orbital escape and trans-Mars maneuver)
TMIS	Trans-Mars Injection System (Stage)	(propulsion and guidance system for TMI)
TMIS-C	TMIS - Cargo	(TMIS for a MCV)
TMIS-P	TMIS - Piloted	(TMIS for a MPV)
TLIS	Trans-Lunar Injection System	(propulsion and guidance system for TLI)
TOD	Tour of Duty	(crew duty time on-station; does not include transport time)
TPS	Thermal Protection System	

Appendix B. Detailed Mass Allocations

These data consist of the following:

Subsystem Masses -- LPV

Subsystem	Mass (kg)
Avionics	363
Structure	1117
Thermal (MLI, other)	206
Aerobrake	1527
Propulsion *	1054
Tankage (w/Meteoroid protect)	1260
Total Dry	5527

*Fixed, incl. engines, TVS, lines, valve

Subsystem Masses -- LCSV

Subsystem	Mass (kg)
Avionics	394
Structure	810
Thermal (MLI, other)	106
Aerobrake	0
Propulsion *	1227
Tankage (w/Meteoroid protect)	347
Total Dry	2884

*Fixed, incl. engines, TVS, lines, valve

Subsystem Masses -- LCV (without LLOX load)

Subsystem	Mass (kg)
Avionics	363
Structure	1117
Thermal (MLI, other)	206
Aerobrake	1527
Propulsion *	1054
Tankage (w/Meteoroid protect)	1260
Total Dry	5527

*Fixed, incl. engines, TVS, lines, valve

Subsystem Masses -- LCL

Subsystem	Mass (kg)
Avionics	394
Structure	1038
Thermal (MLI, other)	106
Aerobrake	0
Propulsion *	1227
Tankage (w/Meteoroid protect)	594
Total Dry	3359

*Fixed, incl. engines, TVS, lines, valve

Subsystem Masses -- LPT

Subsystem	Mass (kg)
Avionics	394
Structure	1038
Thermal (MLI, other)	106
Aerobrake	0
Propulsion *	1227
Tankage (w/Meteoroid protect)	2134
Total Dry	4899

*Fixed, incl. engines, TVS, lines, valve

Subsystem Masses -- LCV (with LLOX load (24t))

Subsystem	Mass (kg)
Avionics	363
Structure	1562
Thermal (MLI, other)	306
Aerobrake	2273
Propulsion *	1054
Tankage (w/Meteoroid protect)	1550
Total Dry	7108

*Fixed, incl. engines, TVS, lines, valve

Mission Summary Sheet

Mission: DAB-FH
L04.Up3c.ChHO.MAb.0D.ChHO.EAb.ETV

Reference mission: DAB-FG *Trajectory file:* A.MASE.BB.TRJ.3/28/89.Ab
Trajectory: Evol. baseline traj: 2004 Mars opp. class/100 d stay
Override(s) in effect: time(s) only.

Mission purpose:
CS-5.0, Cycle-2.6, Mission-1, Ph/DeEV (Cab/MOMV), OR.time, Act.DV, 26Ab
Act PhDeEV prop, MRSR, ETV EOC, TMI:12*RL10X1, TEI:3*Adv.OTV, 2*boiloff

		Mass (t)
M(C1)	Beginning of phase C (Earth escape, TMI)	637.83
M(D1)	Beginning of phase D (transfer to Mars)	232.71
M(D2)	After MCC, DSM, and dumps	218.71
M(E1)	Beginning of phase E (MOC)	218.71
M(Ef)	After achieving capture (& burns and/or drops)	217.64
M(F1)	Beginning of Mars orbital operations	217.64
M(Ff)	End of Mars orbital operations	184.21
M(G1)	Beginning of phase G (Mars escape)	184.21
M(H1)	Beginning of phase H (transfer to Earth)	120.42
M(H2)	After MCC and DSM	115.18
M(I1)	ETV at beginning of EOC	115.18
M(I2)	ETV after EOC	111.83
TMI propellant		367.32
TEI propellant		63.79
all other propellant		27.88
H/O propellant reserve		11.61

Launch date: 5/31/2004. Roundtrip: 1252
Stg/rec/Ab: TMIS-3, TEIS-1, MAb, EAb. Recover: ETV
Artificial gravity: rpm: 4.0; radius: 60.70 ft; struct. aug: 5 %; spin cycles: 4
IMM structure: cyl. mods: 2 partial (100.0 %); disk mods: none; tunnel sect: 5
EPS: Spaceborne: 20 kW solar
Propellant reserve: margin comb: sum; H/O: 11.61 t; Biprop: 0.26 t; other: -0.00 t
Tankage factors: TMIS: 7%; TEIS: 15%
Boiloff: H/O: 7.37 t; other: 0.00 t
Science equip: ISE: 0.30 t; MOSE: 0.15 t;
Consumables: food: 6.59 t; water: 0.00 t; other: 9.22 t

Mass Allocation Report

Mission: DAB-GB
L04.Up3c.ChHO.MAb.0D.ChHO.EAb.ETV

Reference mission: DAB-GB *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Trajectory: Evol. baseline traj: 2004 Mars opp. class/100 d stay
Override(s) in effect: time(s) only.

Mission purpose:
CS-5.0, Cycle-2.6, Mission-1, Ph/DeEV (Cab/MOMV), OR.time, Act.DV, 26Ab
Act PhDeEV prop, MRSR, ETV EOC, TMI:12*RL10X1, TEI:3*Adv.OTV, 2*boiloff

	Mass (t)			
MSS (IMLEO)				637.83
TMIS (Trans-Mars Injection System)			405.12	
Stage 1		125.92		
Propellant (LH2/LOX)	113.32			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
Stage 2		139.60		
Propellant (LH2/LOX)	127.00			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
Stage 3		139.60		
Propellant (LH2/LOX)	127.00			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
MTV			232.71	
MTV (Mars Transfer Vehicle)				232.71
Crew consumables, Mars transfer phase			5.34	
Artificial-g (2 spin-up/downs)			2.36	
Propellant (Stored biprop)		2.36		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
DSM (ETM) (deep space maneuver, Earth-to-Mars)			0.00	
Propellant (LH2/LOX)		0.00		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
MCC (ETM) (mid-course correction)			2.44	
Propellant (LH2/LOX)		2.44		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
RCS (ETM) (reaction control system)			3.86	
Propellant (Stored biprop)		3.86		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
Venus swingby probes(s)			0.00	
MOCS (Mars orbital capture system)			1.06	
Propulsion		1.06		
Propellant (LH2/LOX)	1.06			
Tank(s)	0.00			
Engine(s), avionics+	0.00			
MOV (F1)			217.64	
MOV (Mars Orbiting Vehicle; Phase F1--just after MOC)				217.64
Crew consumables, Mars orbit			0.27	

MOO 1 (Mars orbital operations -1)		4.78	
Propellant (LH2/LOX)	4.78		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
MOO 2		2.17	
Propellant (LH2/LOX)	2.17		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
RCS (MOO)		3.39	
Propellant (Stored biprop)	3.39		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
Satellites		7.50	
RelayComSat(s)	3.00		
MarsSciSat(s)	3.00		
Ph/D teleoperator(s)	1.50		
Teleoperated MRSR		6.00	
MOSE (Mars orbital science equipment)		0.15	
DeEV w/o crew		5.94	
DeEV (recovered)		-5.94	
Earth returnables (received)		-0.10	
Deployed in Mars intermediate orbit		9.27	
AOTPM (Ph/DeEV propulsion system)	9.27		
DeEV simulates Cab/EAb used for the Ph/DeEV too	0.00		
MOV (Ff)		184.21	
MOV (Mars Orbiting Vehicle; Phase Ff--just prior to TEI)			184.21
TEIS (trans-Earth injection system)		63.79	
Propellant (LH2/LOX)	63.79		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
ETV		120.42	
ETV (Earth Transfer Vehicle)			120.42
IMM (interplanetary mission module)		48.05	
Spaceborne external services (power, com, thermal)		1.08	
Crew consumables, transfer to Earth		9.94	
Artificial-g (2 spin-up/downs)		1.21	
Propellant (Stored biprop)	1.21		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
Flyaround		0.00	
Propellant (LH2/LOX)	0.00		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
DSM (MTE)		0.00	
Propellant (LH2/LOX)	0.00		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
MCC (MTE)		1.25	
Propellant (LH2/LOX)	1.25		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
RCS (MTE)		2.77	
Propellant (Stored biprop)	2.00		
Tank(s)	0.64		
Engine(s), avionics+	0.13		
Spacesuits		0.35	
ISE (interplanetary science equipment)		0.30	
Solar/SPE monitoring	0.20		

Astro/Planetary		0.10		
EOCS (Earth orbital capture system)			49.19	
Earth capture Ab		26.00		
Propulsion		23.19		
Propellant (LH2/LOX)	3.35			
Tank(s)	19.30			
Engine(s), avionics+	0.54			
DeEV (recovered)			5.94	
Crew+returnables			0.34	
IMM (Interplanetary Mission Modules)				48.05
Cylindrical Module(s)			34.00	
Disk Module(s)			0.00	
Tunnel(s)			1.50	
Tanks for crew consumables			0.78	
Artificial-g structure			2.27	
Resource Nodes (docking, prox ops)			3.00	
Airlock(s) (AL)			0.30	
Hyperbaric airlock(s) (HAL)			0.70	
Radiation shelter shielding			2.00	
Life support system (LSS)			1.40	
Data management system (DMS)			0.30	
Internal Com/EPS/TCS			1.80	
Spaceborne External Services				1.08
Electrical power system (EPS), external			0.28	
Thermal control system (TCS), external			0.30	
Communications system, external			0.50	

Propulsion System Summary Report

Mission: DAB-FH
L04.Up3c.ChHO.MAb.0D.ChHO.EAb.ETV

Reference mission: DAB-FG *Trajectory file:* A.MASE.BB.TRJ.3/28/89.Ab
Trajectory: Evol. baseline traj: 2004 Mars opp. class/100 d stay
Override(s) in effect: time(s) only.

Mission purpose:
CS-5.0, Cycle-2.6, Mission-1, Ph/DeEV (Cab/MOMV), OR.time, Act.DV, 26Ab
Act PhDeEV prop, MRSR, ETV EOC, TMI:12*RL10X1, TEI:3*Adv.OTV, 2*boiloff

Element	Delta V (km/s)	Prop. Sys. #	Propellant	Isp (lbf-s/lbm)	Mi/Mf	Total thrust (lbf)
TMI stage 1	0.885	1	LH2/LOX	471	1.211	532000
TMI stage 2	1.289	2	LH2/LOX	471	1.322	532000
TMI stage 3	1.885	3	LH2/LOX	471	1.504	532000
ETM DSM	0.000	5	LH2/LOX	480	1.000	22526
Art-g ETM	0.031	17	Stored biprop	311	1.010	3000
ETM MCC	0.050	5	LH2/LOX	480	1.011	22526
RCS ETM	0.050	17	Stored biprop	311	1.017	3000
MOC post A/C peri. raise	0.020	5	LH2/LOX	480	1.004	22526
MOO 1	0.100	5	LH2/LOX	480	1.021	22526
MOO 2	0.050	5	LH2/LOX	480	1.011	22526
RCS MOO	0.050	17	Stored biprop	311	1.017	3000
TEI	1.770	5	LH2/LOX	480	1.456	22526
MTE DSM	0.000	5	LH2/LOX	480	1.000	22526
Art-g MTE	0.031	17	Stored biprop	311	1.010	3000
MTE MCC	0.050	5	LH2/LOX	480	1.011	22526
RCS MTE	0.050	17	Stored biprop	311	1.017	3000
ETV post A/C peri. raise	0.136	5	LH2/LOX	480	1.029	22526

◊ Indicates that value overrides that in engines/tanks data base.

Crew Consumables* Report

Mission: DAB-FH
L04.Up3c.ChHO.MAb.0D.ChHO.EAb.ETV

Reference mission: DAB-FH *Trajectory file:* A.MASE.BB.TRJ.3/28/89.Ab
Trajectory: Evol. baseline traj: 2004 Mars opp. class/100 d stay
Override(s) in effect: time(s) only.

Mission purpose:
CS-5.0, Cycle-2.6, Mission-1, Ph/DeEV (Cab/MOMV), OR.time, Act.DV, 26Ab
Act PhDeEV prop, MRSR, ETV EOC, TMI:12*RL10X1, TEI:3*Adv.OTV, 2*boiloff

Crew composition: Nominal U. S. gender-mixed crew

Period	Mission phase	# of crew	Time	Person-days	Margin	Total mass (t)
LEO Checkout	A	3	21 day	63	15 %	0.26
MTV	D	3	430 day	1290	15 %	5.34
MOV	F		22 day	66	15 %	0.27
ETV	H	3	800 day	2400	15 %	9.94
					Total (incl. margin)	15.81
					Total (w/o margin)	13.75

Total supply = 10.46 person-years = 3819 person-days
Average supply = 4.14 kg/person-day

Water: crew prod. = 3.4 kg/p-d; hygiene = 8.0 kg/p-d; recycling efficiency = 90.0 %

Consumables Baseline (nominal U. S. gender-mixed crew, kg/person-day):

	Food	Pot. Water	Other	Total
Spaceborne	1.5	1.0	2.1	4.6
Surface	1.5	2.0	2.0	5.5
MAV, ECCV	1.5	2.0	2.0	5.5
Mission Totals (t) (includes LEO checkout)				
Consumption	5.73	0.00	8.02	13.75
Initial Storage	6.59	0.00	9.22	15.81

* Consumables includes LSS + Food
(a) To provide interplanetary safe-haven capability.

Propulsion Engines Report

Mission: DAB-FH
L04.Up3c.ChHO.MAb.0D.ChHO.EAb.ETV

Reference mission: DAB-FH *Trajectory file:* A.MASE.BB.TRJ.3/28/89.Ab
Trajectory: Evol. baseline traj: 2004 Mars opp. class/100 d stay
Override(s) in effect: time(s) only.

Mission purpose:
CS-5.0, Cycle-2.6, Mission-1, Ph/DeEV (Cab/MOMV), OR.time, Act.DV, 26Ab
Act PhDeEV prop, MRSR, ETV EOC, TMI:12*RL10X1, TEI:3*Adv.OTV, 2*boiloff

Element	Propellant	Engine Ident.	Engine name	Rev.	Isp (lbf-s/lbm)	Mass per engine (kg
TMI stage 1	LH2/LOX	14	SSME, adv, 306:1	2	471	3560.2
TMI stage 2	LH2/LOX	14	SSME, adv, 306:1	2	471	3560.2
TMI stage 3	LH2/LOX	14	SSME, adv, 306:1	2	471	3560.2
ETM DSM	LH2/LOX	21	Advanced OTV	2	480	163.0
Art-g ETM	Stored bipro	47	Marquardt R-4D	2	311	3.8
ETM MCC	LH2/LOX	21	Advanced OTV	2	480	163.0
RCS ETM	Stored bipro	47	Marquardt R-4D	2	311	3.8
MOC post A/C peri. raise	LH2/LOX	21	Advanced OTV	2	480	163.0
MOO 1	LH2/LOX	21	Advanced OTV	2	480	163.0
MOO 2	LH2/LOX	21	Advanced OTV	2	480	163.0
RCS MOO	Stored bipro	47	Marquardt R-4D	2	311	3.8
TEI	LH2/LOX	21	Advanced OTV	2	480	163.0
MTE DSM	LH2/LOX	21	Advanced OTV	2	480	163.0
Art-g MTE	Stored bipro	47	Marquardt R-4D	2	311	3.8
MTE MCC	LH2/LOX	21	Advanced OTV	2	480	163.0
RCS MTE	Stored bipro	47	Marquardt R-4D	2	311	3.8
ETV post A/C peri. raise	LH2/LOX	21	Advanced OTV	2	480	163.0

◊ Indicates that value overrides that in engines/tanks data base.

Propulsion Tanks Report

Mission: DAB-FH
L04.Up3c.ChHO.MAb.0D.ChHO.EAb.ETV

Reference mission: DAB-FH *Trajectory file:* A.MASE.BB.TRJ.3/28/89.Ab
Trajectory: Evol. baseline traj: 2004 Mars opp. class/100 d stay
Override(s) in effect: time(s) only.

Mission purpose:
CS-5.0, Cycle-2.6, Mission-1, Ph/DeEV (Cab/MOMV), OR.time, Act.DV, 26Ab
Act PhDeEV prop, MRSR, ETV EOC, TMI:12*RL10X1, TEI:3*Adv.OTV, 2*boiloff

Element	Propellant	Tank Ident.	Tank name	Rev.	Tankage Fact.(%)	Cap.(t)	Margins(%) Bulk	dV	Isp
TMI stage 1	LH2/LOX	9	TMI,med-tf,med-bo	4	7.0	127.0	1.0	0.0	1.0
TMI stage 2	LH2/LOX	9	TMI,med-tf,med-bo	4	7.0	127.0	1.0	0.0	1.0
TMI stage 3	LH2/LOX	9	TMI,med-tf,med-bo	4	7.0	127.0	1.0	0.0	1.0
ETM DSM	LH2/LOX	24	Fixed TEI, 19.305	5	15.0	128.7	1.0	0.0	1.0
Art-g ETM	Stored bipro	46	Small biprop (MCC	2	5.0	Rubber	1.0	0.0	1.0
ETM MCC	LH2/LOX	24	Fixed TEI, 19.305	5	15.0	128.7	1.0	0.0	1.0
RCS ETM	Stored bipro	46	Small biprop (MCC	2	5.0	Rubber	1.0	0.0	1.0
MOC post A/C peri. raise	LH2/LOX	24	Fixed TEI, 19.305	5	15.0	128.7	1.0	0.0	1.0
MOO 1	LH2/LOX	24	Fixed TEI, 19.305	5	15.0	128.7	1.0	0.0	1.0
MOO 2	LH2/LOX	24	Fixed TEI, 19.305	5	15.0	128.7	1.0	0.0	1.0
RCS MOO	Stored bipro	46	Small biprop (MCC	2	5.0	Rubber	1.0	0.0	1.0
TEI	LH2/LOX	24	Fixed TEI, 19.305	5	15.0	128.7	2.0	0.0	1.0
MTE DSM	LH2/LOX	24	Fixed TEI, 19.305	5	15.0	128.7	1.0	0.0	1.0
Art-g MTE	Stored bipro	46	Small biprop (MCC	2	5.0	Rubber	1.0	0.0	1.0
MTE MCC	LH2/LOX	24	Fixed TEI, 19.305	5	15.0	128.7	1.0	0.0	1.0
RCS MTE	Stored bipro	46	Small biprop (MCC	2	5.0	Rubber	1.0	0.0	1.0
ETV post A/C peri. raise	LH2/LOX	24	Fixed TEI, 19.305	5	15.0	128.7	1.0	0.0	1.0

* Combination of reserves due to margins: sum
** Interplanetary trajectory boiloff thermal factor (relative to 1 A. U.):
Human outbound = 120.0 % Human inbound = 100.0 %

◊ Indicates that value overrides that in engines/tanks data base.

Astrodynamics Report

Mission: DAB-FH
L04.Up3c.ChHO.MAb.0D.ChHO.EAb.ETV

Reference mission: DAB-FG *Trajectory file:* A.MASE.BB.TRJ.3/28/89.Ab
Trajectory: Evol. baseline traj: 2004 Mars opp. class/100 d stay
Override(s) in effect: time(s) only.

Mission purpose:
CS-5.0, Cycle-2.6, Mission-1, Ph/DeEV (Cab/MOMV), OR.time, Act.DV, 26Ab
Act PhDeEV prop, MRSR, ETV EOC, TMI:12*RL10X1, TEI:3*Adv.OTV, 2*boiloff

Trajectory Type: Evol. baseline traj: 2004 Mars opp. class/100 d stay

	Date	Apoapsis (km)	Periapsis (km)	Inclination (deg)	Entry Vel. (km/s)	C3 (km^2/s^2)
Earth departure (TMI)	5/31/2004	500	500	28.50		20.27
Venus swingby	11/17/2004					
Mars arrival (MOC)	4/10/2005	18000	250	76.35	9.060	57.65
Mars departure (TEI)	7/19/2005	18000	250	76.35		15.60
Earth arrival (EOC)	1/13/2006	500	500	28.50	11.599	11.62

Duration

	Trajectory			Override		
	Sols	Days	Months	Days	Months	
Marsbound (ETM)		314.5	10.33	430.0	14.13	Use override time
Mars orbit	97.2	100.0	3.29	22.0	0.72	Use override time
Earthbound (MTE)		177.8	5.84	800.0	26.28	Use override time
Total trip		592.2	19.46	1252.0	41.13	

Delta V Summary

	Delta V (km/s)		
Item	Trajectory	Override	
TMI	4.059	4.400	Use trajectory delta V
ETM DSM	0.000	0.000	Use trajectory delta V
Art-g ETM	0.031		
ETM MCC	0.050		
RCS ETM	0.050		
MOC post A/C periapsis raise	0.020	0.020	Use trajectory delta V
MOO 1	0.100		
MOO 2	0.050		
RCS MOO	0.050		
TEI	1.770	2.650	Use trajectory delta V
MTE DSM	0.000	0.000	Use trajectory delta V
Art-g MTE	0.031		
MTE MCC	0.050		
RCS MTE	0.050		
ETV post A/C periapsis raise	0.136	0.040	Use trajectory delta V

Mission Summary Sheet

Mission: DAB-FF
L05.Up5c.ChHO.MAb.0D.ChHO.EAb.ETV

Reference mission: DAB-FE *Trajectory file:* A.MASE.BB.TRJ.3/28/89.Ab
Trajectory: Evol. baseline traj: 2005 Mars multi-rev class/200 d stay
Override(s) in effect: time(s) only.

Mission purpose:
CS-5.0, Cycle-2.6, Mission-2, PhEV: LAM and MTOS, DeEV: Cab/ECCV
All-Up 2005 M-rev, OR.time, Act.DV, 26t Ab, Fix TEI, 20.08t landed cargo

		Mass (t)
M(C1)	Beginning of phase C (Earth escape, TMI)	687.16
M(D1)	Beginning of phase D (transfer to Mars)	253.08
M(D2)	After MCC, DSM, and dumps	234.30
M(E1)	Beginning of phase E (MOC)	234.30
M(Ef)	After achieving capture (& burns and/or drops)	233.58
M(F1)	Beginning of Mars orbital operations	233.58
M(Ff)	End of Mars orbital operations	162.51
M(G1)	Beginning of phase G (Mars escape)	162.51
M(H1)	Beginning of phase H (transfer to Earth)	128.61
M(H2)	After MCC and DSM	122.89
M(I1)	ETV at beginning of EOC	122.89
M(I2)	ETV after EOC	119.32
TMI propellant		383.68
TEI propellant		33.90
all other propellant		26.96
H/O propellant reserve		10.67

Launch date: 8/22/2005. Roundtrip: 1252
Stg/rec/Ab: TMIS-4, TEIS-1, MAb, EAb. Recover: ETV
Artificial gravity: rpm: 4.0; radius: 60.70 ft; struct. aug: 5 %; spin cycles: 4
IMM structure: cyl. mods: 2 partial (100.0 %); disk mods: none; tunnel sect: 5
EPS: Spaceborne: 20 kW solar
Propellant reserve: margin comb: sum; H/O: 10.67 t; Biprop: 0.28 t; other: 0.00 t
Tankage factors: TMIS: 7%; TEIS: 15%
Boiloff: H/O: 3.49 t; other: 0.00 t
Science equip: ISE: 0.30 t; MOSE: 0.15 t;
Consumables: food: 10.98 t; water: 0.00 t; other: 15.37 t

Mass Allocation Report

Mission: DAB-GC
L05.Up5c.ChHO.MAb.0D.ChHO.EAb.ETV

Reference mission: DAB-GC *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Trajectory: Evol. baseline traj: 2005 Mars multi-rev class/200 d stay
Override(s) in effect: time(s) only.

Mission purpose:
CS-5.0, Cycle-2.6, Mission-2, PhEV: LAM and MTOS, DeEV: Cab/ECCV
All-Up 2005 M-rev, OR.time, Act.DV, 26t Ab, Fix TEI, 20.08t landed cargo

	Mass (t)			
MSS (IMLEO)				687.16
TMIS (Trans-Mars Injection System)			434.08	
Stage 1		15.28		
Propellant (LH2/LOX)	2.68			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
Stage 2		139.60		
Propellant (LH2/LOX)	127.00			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
Stage 3		139.60		
Propellant (LH2/LOX)	127.00			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
Stage 4		139.60		
Propellant (LH2/LOX)	127.00			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
MTV			253.08	
MTV (Mars Transfer Vehicle)				253.08
Crew consumables, Mars transfer phase			9.36	
Artificial-g (2 spin-up/downs)			2.58	
Propellant (Stored biprop)		2.58		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
DSM (ETM) (deep space maneuver, Earth-to-Mars)			0.00	
Propellant (LH2/LOX)		0.00		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
MCC (ETM) (mid-course correction)			2.65	
Propellant (LH2/LOX)		2.65		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
RCS (ETM) (reaction control system)			4.20	
Propellant (Stored biprop)		4.20		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
MOCS (Mars orbital capture system)			0.72	
Propulsion		0.72		
Propellant (LH2/LOX)	0.72			
Tank(s)	0.00			
Engine(s), avionics+	0.00			
MOV (F1)			233.58	

MOV (Mars Orbiting Vehicle; Phase F1--just after MOC)			233.58
Crew consumables, Mars orbit		0.00	
MOO 1 (Mars orbital operations -1)		2.71	
Propellant (LH2/LOX)	2.71		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
MOO 2		1.87	
Propellant (LH2/LOX)	1.87		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
RCS (MOO)		3.87	
Propellant (Stored biprop)	3.87		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
Satellites		0.00	
RelayComSat(s)	0.00		
MarsSciSat(s)	0.00		
Ph/D teleoperator(s)	0.00		
Teleoperated MRSR		0.00	
MOSE (Mars orbital science equipment)		0.15	
DeEV w/o crew		5.94	
DeEV (recovered)		-5.94	
Earth returnables (received)		-0.10	
Deployed in Mars intermediate orbit		62.55	
LAPM, wet (lander part of MDV)	27.24		
AOTPM, wet (ascent part of MDV)	15.23		
Crew Cab (bookkept under DeEV)	0.00		
Mars Landed Mission Module	11.52		
Mars Landed Operational Equipment	0.80		
Mars Landed Expendables	7.76		
MOV (Ff)		162.51	

MOV (Mars Orbiting Vehicle; Phase Ff--just prior to TEI)			162.51
TEIS (trans-Earth injection system)		33.90	
Propellant (LH2/LOX)	33.90		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
ETV		128.61	

ETV (Earth Transfer Vehicle)			128.61
IMM (interplanetary mission module)		48.61	
Spaceborne external services (power, com, thermal)		1.08	
Crew consumables, transfer to Earth		16.56	
Artificial-g (2 spin-up/downs)		1.57	
Propellant (Stored biprop)	1.30		
Tank(s)	0.19		
Engine(s), avionics+	0.07		
Flyaround		0.00	
Propellant (LH2/LOX)	0.00		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
DSM (MTE)		0.00	
Propellant (LH2/LOX)	0.00		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
MCC (MTE)		1.34	
Propellant (LH2/LOX)	1.34		
Tank(s)	0.00		
Engine(s), avionics+	0.00		

RCS (MTE)			2.81	
Propellant (Stored biprop)		2.13		
Tank(s)		0.51		
Engine(s), avionics+		0.17		
Spacesuits			0.49	
ISE (interplanetary science equipment)			0.30	
Solar/SPE monitoring		0.20		
Astro/Planetary		0.10		
EOCS (Earth orbital capture system)			49.42	
Earth capture Ab		26.00		
Propulsion		23.42		
Propellant (LH2/LOX)	3.57			
Tank(s)	19.30			
Engine(s), avionics+	0.54			
DeEV (recovered)			5.94	
Crew+returnables			0.50	
IMM (Interplanetary Mission Modules)				48.61
Cylindrical Module(s)			34.00	
Disk Module(s)			0.00	
Tunnel(s)			1.50	
Tanks for crew consumables			1.30	
Artificial-g structure			2.31	
Resource Nodes (docking, prox ops)			3.00	
Airlock(s) (AL)			0.30	
Hyperbaric airlock(s) (HAL)			0.70	
Radiation shelter shielding			2.00	
Life support system (LSS)			1.40	
Data management system (DMS)			0.30	
Internal Com/EPS/TCS			1.80	
Spaceborne External Services				1.08
Electrical power system (EPS), external			0.28	
Thermal control system (TCS), external			0.30	
Communications system, external			0.50	

Mission Summary Sheet

Mission: DAB-FJ
L07.Hum3c.ChHO.MAb.0D.ChHO
L07.Car.ChHO.MAb

Reference mission: DAB-FB *Trajectory file:* A.MASE.BB.TRJ.3/28/89.Ab
Human traj: Evol. baseline traj: 2007 Mars one-way
Cargo traj: Evol. baseline traj: 2007 Mars one-way
Override(s) in effect: time(s) & delta V(s).

Mission purpose:
CS-5.0, Cycle-2.6, Mission-3, Cargo to Gateway, ISPP (25t), tether (75t), 10t VCF, 50t surf.cargo and Lander (LAPM 27t),

Total IMLEO = 1757.38 t

		Human mission mass (t)	Cargo mission mass (t)
M(C1)	Beginning of phase C (Earth escape, TMI)	1032.06	725.32
M(D1)	Beginning of phase D (transfer to Mars)	361.16	281.33
M(D2)	After MCC, DSM, and dumps	312.64	273.68
M(E1)	Beginning of phase E (MOC)	312.64	273.68
M(Ef)	After achieving capture (& burns and/or drops)	270.36	212.35
M(F1)	Beginning of Mars orbital operations	270.36	212.35
M(Ff)	End of Mars orbital operations	25.56	1.38
M(G1)	Beginning of phase G (Mars escape)	25.56	
M(H1)	Beginning of phase H (transfer to Earth)	25.56	
M(H2)	After MCC and DSM	22.49	
TMI propellant		635.33	393.59
TEI propellant		0.00	
all other propellant		40.95	46.50
H/O propellant reserve		22.21	10.16

Launch dates: Cargo: 10/5/2007. Human: 10/5/2007. Hum Roundtrip: 876.75 d
Stg/rec/Ab: TMIS-4, TEIS-1, MAb. Recover: TMIS
Artificial gravity: rpm: 2.0; radius: 280.00 ft; struct. aug: 5 %; spin cycles: 4
IMM structure: cyl. mods: 2 partial (40.0 %); disk mods: none; tunnel sect: 1
EPS: Spaceborne: 20 kW solar
Propellant reserve: margin comb: sum; H/O: 32.37 t; Biprop: 0.93 t; other: 0.00 t
Tankage factors: TMIS: 7%; TEIS: 15%
Boiloff: H/O: 4.42 t; other: 0.00 t
Science equip: ISE: 0.30 t; MOSE: 0.15 t;
Consumables: food: 4.65 t; water: 0.00 t; other: 6.50 t

Cargo Mission
Mass Allocation Report

Mission: DAB-GD
L07.Hum3c.ChHO.MAb.0D.ChHO
L07.Car.ChHO.MAb

Reference mission: DAB-FJ *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Human traj: Evol. baseline traj: 2007 Mars one-way
Cargo traj: Evol. baseline traj: 2007 Mars one-way
Override(s) in effect: time(s) & delta V(s).

Mission purpose:
CS-5.0, Cycle-2.6, Mission-3, Cargo to Gateway, ISPP (25t), tether (75t), 10t VCF, 50t surf.cargo and Lander (LAPM 27t),

	Mass (t)			
MCV (IMLEO)				725.32
TMIS			444.00	
Stage 1		25.19		
Propellant (LH2/LOX)	12.59			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
Stage 2		139.60		
Propellant (LH2/LOX)	127.00			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
Stage 3		139.60		
Propellant (LH2/LOX)	127.00			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
Stage 4		139.60		
Propellant (LH2/LOX)	127.00			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
MTV			281.33	
MTV				281.33
MCC			2.98	
Propellant (LH2/LOX)		2.98		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
RCS (ETM)			4.67	
Propellant (Stored biprop)		4.67		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
MOCS			61.33	
Mars capture Ab		26.00		
Propulsion		35.33		
Propellant (LH2/LOX)	35.33			
Tank(s)	0.00			
Engine(s), avionics+	0.00			
MOV (F1)			212.35	
MOV (F1)				212.35
MOO			19.85	
Propellant (LH2/LOX)		0.00		

Tank(s)		19.30	
Engine(s), avionics+		0.54	
RCS (MOO)			4.10
Propellant (Stored biprop)		3.52	
Tank(s)		0.41	
Engine(s), avionics+		0.17	
Payload			187.02
Satellites		0.00	
RelayComSat(s)	0.00		
MarsSciSat(s)	0.00		
Ph/D teleoperator(s)	0.00		
Teleoperated MRSR		0.00	
MOSE		0.00	
ISE		0.00	
Solar/SPE monitoring	0.00		
Astro/Planetary	0.00		
Deployed in Mars intermediate orbit		187.02	
Mars Cargo Lander (wet, no payload)	27.02		
MCL payload	50.00		
In Situ Propellant Production Plant	25.00		
Vehicle Changeout Facility	10.00		
Tether System (line: 50t, Reel&Motor: 20t, Tower: 5t)	75.00		
Structure			0.50
Support Services			0.88
Data management system (DMS)		0.05	
Electrical power system (EPS)		0.18	
Thermal control system (TCS)		0.15	
Communications system		0.50	

Mission Summary Sheet

Mission: DAB-ER
L09.Up5c.ChHO.MAb.0D.ChHO.EAb.ETV

Reference mission: DAB-EM *Trajectory file:* A.MASE.BB.TRJ.3/28/89.Ab
Trajectory: Evol. baseline traj: 2009 Mars conjunction class
Override(s) in effect: time(s) & delta V(s).

Mission purpose:
CS-5.0, Cycle-2.6, Mission-4, MCSV (LAM+APM+Cab), 10t surf.cargo,
OR.time, Act.DV, PhEV: MCSV, DeEV: ECCV (Cab/EAb), Reuse of M-1 MPV

		Mass (t)
M(C1)	Beginning of phase C (Earth escape, TMI)	510.39
M(D1)	Beginning of phase D (transfer to Mars)	195.13
M(D2)	After MCC, DSM, and dumps	178.50
M(E1)	Beginning of phase E (MOC)	178.50
M(Ef)	After achieving capture (& burns and/or drops)	178.50
M(F1)	Beginning of Mars orbital operations	178.50
M(Ff)	End of Mars orbital operations	123.44
M(G1)	Beginning of phase G (Mars escape)	123.44
M(H1)	Beginning of phase H (transfer to Earth)	123.44
M(H2)	After MCC and DSM	119.33
M(I1)	ETV at beginning of EOC	119.33
M(I2)	ETV after EOC	119.32

TMI propellant	277.46
TEI propellant	0.00
all other propellant	34.52
H/O propellant reserve	7.61

Launch date: 10/15/2009. Roundtrip: 1252
Stg/rec/Ab: TMIS-3, TEIS-1, MAb, EAb. Recover: ETV
Artificial gravity: rpm: 4.0; radius: 60.70 ft; struct. aug: 5 %; spin cycles: 4
IMM structure: cyl. mods: 2 partial (100.0 %); disk mods: none; tunnel sect: 5
EPS: Spaceborne: 20 kW solar
Propellant reserve: margin comb: sum; H/O: 7.61 t; Biprop: 0.23 t; other: 0.00 t
Tankage factors: TMIS: 7%; TEIS: 15%
Boiloff: H/O: 1.87 t; other: 0.00 t
Science equip: ISE: 0.30 t; MOSE: 0.15 t;
Consumables: food: 10.98 t; water: 0.00 t; other: 15.37 t

Mass Allocation Report

Mission: DAB-GE
L09.Up5c.ChHO.MAb.0D.ChHO.EAb.ETV

Reference mission: DAB-ER *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Trajectory: Evol. baseline traj: 2009 Mars conjunction class
Override(s) in effect: time(s) & delta V(s).

Mission purpose:
CS-5.0, Cycle-2.6, Mission-4, MCSV (LAM+APM+Cab), 10t surf.cargo, OR.time, Act.DV, PhEV: MCSV, DeEV: ECCV (Cab/EAb), Reuse of M-1 MPV

	Mass (t)			
MSS (IMLEO)				510.39
TMIS (Trans-Mars Injection System)			315.26	
Stage 1		36.06		
Propellant (LH2/LOX)	23.46			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
Stage 2		139.60		
Propellant (LH2/LOX)	127.00			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
Stage 3		139.60		
Propellant (LH2/LOX)	127.00			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
MTV			195.13	
MTV (Mars Transfer Vehicle)				195.13
Crew consumables, Mars transfer phase			9.36	
Artificial-g (2 spin-up/downs)			1.99	
Propellant (Stored biprop)		1.99		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
DSM (ETM) (deep space maneuver, Earth-to-Mars)			0.00	
Propellant (LH2/LOX)		0.00		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
MCC (ETM) (mid-course correction)			2.05	
Propellant (LH2/LOX)		2.05		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
RCS (ETM) (reaction control system)			3.24	
Propellant (Stored biprop)		3.24		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
MOCS (Mars orbital capture system)			0.00	
Propulsion		0.00		
Propellant (LH2/LOX)	0.00			
Tank(s)	0.00			
Engine(s), avionics+	0.00			
MOV (F1)			178.50	
MOV (Mars Orbiting Vehicle; Phase F1--just after MOC)				178.50
Crew consumables, Mars orbit			0.00	
MOO 1 (Mars orbital operations -1)			20.99	

Propellant (LH2/LOX)	20.99		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
MOO 2		0.00	
Propellant (LH2/LOX)	0.00		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
RCS (MOO)		2.96	
Propellant (Stored biprop)	2.96		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
Satellites		0.00	
RelayComSat(s)	0.00		
MarsSciSat(s)	0.00		
Ph/D teleoperator(s)	0.00		
Teleoperated MRSR		0.00	
MOSE (Mars orbital science equipment)		0.15	
DeEV w/o crew		5.94	
DeEV (recovered)		-5.94	
Earth returnables (received)		-0.10	
Deployed in Mars intermediate orbit		31.06	
Dry LAPM (part of MCSV)	13.24		
Dry APM (part of MCSV)	2.88		
Crew Cab (part of MCSV)	4.94		
Cargo for MCSV destine to Mars' surface	10.00		
MOV (Ff)		123.44	
MOV (Mars Orbiting Vehicle; Phase Ff--just prior to TEI)			123.44
TEIS (trans-Earth injection system)		0.00	
Propellant (LH2/LOX)	0.00		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
ETV		123.44	
ETV (Earth Transfer Vehicle)			123.44
IMM (interplanetary mission module)		48.61	
Spaceborne external services (power, com, thermal)		1.08	
Crew consumables, transfer to Earth		16.56	
Artificial-g (2 spin-up/downs)		1.49	
Propellant (Stored biprop)	1.25		
Tank(s)	0.16		
Engine(s), avionics+	0.07		
Flyaround		0.00	
Propellant (LH2/LOX)	0.00		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
DSM (MTE)		0.00	
Propellant (LH2/LOX)	0.00		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
MCC (MTE)		0.00	
Propellant (LH2/LOX)	0.00		
Tank(s)	0.00		
Engine(s), avionics+	0.00		
RCS (MTE)		2.63	
Propellant (Stored biprop)	2.05		
Tank(s)	0.41		
Engine(s), avionics+	0.17		
Spacesuits		0.49	
ISE (interplanetary science equipment)		0.30	

Solar/SPE monitoring		0.20		
Astro/Planetary		0.10		
EOCS (Earth orbital capture system)			45.85	
Earth capture Ab		26.00		
Propulsion		19.85		
Propellant (LH2/LOX)	0.00			
Tank(s)	19.30			
Engine(s), avionics+	0.54			
DeEV (recovered)			5.94	
Crew+returnables			0.50	
IMM (Interplanetary Mission Modules)				48.61
Cylindrical Module(s)			34.00	
Disk Module(s)			0.00	
Tunnel(s)			1.50	
Tanks for crew consumables			1.30	
Artificial-g structure			2.31	
Resource Nodes (docking, prox ops)			3.00	
Airlock(s) (AL)			0.30	
Hyperbaric airlock(s) (HAL)			0.70	
Radiation shelter shielding			2.00	
Life support system (LSS)			1.40	
Data management system (DMS)			0.30	
Internal Com/EPS/TCS			1.80	
Spaceborne External Services				1.08
Electrical power system (EPS), external			0.28	
Thermal control system (TCS), external			0.30	
Communications system, external			0.50	

Mission Summary Sheet

Mission: DAB-EP
L11.Up5c.ChHO.MAb.0D.ChHO.EAb.ETV

Reference mission: DAB-DZ *Trajectory file:* A.MASE.BB.TRJ.3/28/89.Ab
Trajectory: Evol. baseline traj: 2011 conjunction class
Override(s) in effect: time(s) & delta V(s).

Mission purpose:
CS-5.0, Cycle-2.6, Mission-5, Uses gateway, no MCSV, OR.time, Act.DV
All-Up 2011 Cn, ECCV (Cab/EAb), 10t surface cargo

		Mass (t)
M(C1)	Beginning of phase C (Earth escape, TMI)	428.03
M(D1)	Beginning of phase D (transfer to Mars)	169.78
M(D2)	After MCC, DSM, and dumps	154.10
M(E1)	Beginning of phase E (MOC)	154.10
M(Ef)	After achieving capture (& burns and/or drops)	154.10
M(F1)	Beginning of Mars orbital operations	154.10
M(Ff)	End of Mars orbital operations	123.37
M(G1)	Beginning of phase G (Mars escape)	123.37
M(H1)	Beginning of phase H (transfer to Earth)	123.37
M(H2)	After MCC and DSM	119.31
M(I1)	ETV at beginning of EOC	119.31
M(I2)	ETV after EOC	119.31
TMI propellant		233.05
TEI propellant		0.00
all other propellant		30.30
H/O propellant reserve		6.64

Launch date: 11/20/2011. Roundtrip: 1252
Stg/rec/Ab: TMIS-2, TEIS-1, MAb, EAb. Recover: ETV
Artificial gravity: rpm: 4.0; radius: 60.70 ft; struct. aug: 5 %; spin cycles: 4
IMM structure: cyl. mods: 2 partial (100.0 %); disk mods: none; tunnel sect: 5
EPS: Spaceborne: 20 kW solar
Propellant reserve: margin comb: sum; H/O: 6.64 t; Biprop: 0.21 t; other: 0.00 t
Tankage factors: TMIS: 7%; TEIS: 15%
Boiloff: H/O: 1.61 t; other: 0.00 t
Science equip: ISE: 0.30 t; MOSE: 0.15 t;
Consumables: food: 10.98 t; water: 0.00 t; other: 15.37 t

Mass Allocation Report

Mission: DAB-GF
L11.Up5c.ChHO.MAb.0D.ChHO.EAb.ETV

Reference mission: DAB-EP *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Trajectory: Evol. baseline traj: 2011 conjunction class
Override(s) in effect: time(s) & delta V(s).

Mission purpose:
CS-5.0, Cycle-2.6, Mission-5, Uses gateway, no MCSV, OR.time, Act.DV
All-Up 2011 Cn, ECCV (Cab/EAb), 10t surface cargo

	Mass (t)			
MSS (IMLEO)				428.03
TMIS (Trans-Mars Injection System)			258.25	
Stage 1		118.65		
Propellant (LH2/LOX)	106.05			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
Stage 2		139.60		
Propellant (LH2/LOX)	127.00			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
MTV			169.78	

MTV (Mars Transfer Vehicle)				169.78
Crew consumables, Mars transfer phase			9.36	
Artificial-g (2 spin-up/downs)			1.73	
Propellant (Stored biprop)		1.73		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
DSM (ETM) (deep space maneuver, Earth-to-Mars)			0.00	
Propellant (LH2/LOX)		0.00		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
MCC (ETM) (mid-course correction)			1.78	
Propellant (LH2/LOX)		1.78		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
RCS (ETM) (reaction control system)			2.82	
Propellant (Stored biprop)		2.82		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
MOCS (Mars orbital capture system)			0.00	
Propulsion		0.00		
Propellant (LH2/LOX)	0.00			
Tank(s)	0.00			
Engine(s), avionics+	0.00			
MOV (F1)			154.10	

MOV (Mars Orbiting Vehicle; Phase F1--just after MOC)				154.10
Crew consumables, Mars orbit			0.00	
MOO 1 (Mars orbital operations -1)			18.12	
Propellant (LH2/LOX)		18.12		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
MOO 2			0.00	

Propellant (LH2/LOX)		0.00		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
RCS (MOO)			2.56	
Propellant (Stored biprop)		2.56		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
Satellites			0.00	
RelayComSat(s)		0.00		
MarsSciSat(s)		0.00		
Ph/D teleoperator(s)		0.00		
Teleoperated MRSR			0.00	
MOSE (Mars orbital science equipment)			0.15	
DeEV w/o crew			5.94	
DeEV (recovered)			-5.94	
Earth returnables (received)			-0.10	
Deployed in Mars intermediate orbit			10.00	
Cargo to be transferred to MCSV		10.00		
MOV (Ff)			123.37	
MOV (Mars Orbiting Vehicle; Phase Ff--just prior to TEI)				123.37
TEIS (trans-Earth injection system)			0.00	
Propellant (LH2/LOX)		0.00		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
ETV			123.37	
ETV (Earth Transfer Vehicle)				123.37
IMM (interplanetary mission module)			48.60	
Spaceborne external services (power, com, thermal)			1.08	
Crew consumables, transfer to Earth			16.56	
Artificial-g (2 spin-up/downs)			1.47	
Propellant (Stored biprop)		1.25		
Tank(s)		0.15		
Engine(s), avionics+		0.07		
Flyaround			0.00	
Propellant (LH2/LOX)		0.00		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
DSM (MTE)			0.00	
Propellant (LH2/LOX)		0.00		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
MCC (MTE)			0.00	
Propellant (LH2/LOX)		0.00		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
RCS (MTE)			2.59	
Propellant (Stored biprop)		2.05		
Tank(s)		0.37		
Engine(s), avionics+		0.17		
Spacesuits			0.49	
ISE (interplanetary science equipment)			0.30	
Solar/SPE monitoring		0.20		
Astro/Planetary		0.10		
EOCS (Earth orbital capture system)			45.85	
Earth capture Ab		26.00		
Propulsion		19.85		
Propellant (LH2/LOX)	0.00			
Tank(s)	19.30			

Engine(s), avionics+	0.54		
DeEV (recovered)		5.94	
Crew+returnables		0.50	
IMM (Interplanetary Mission Modules)			48.60
Cylindrical Module(s)		34.00	
Disk Module(s)		0.00	
Tunnel(s)		1.50	
Tanks for crew consumables		1.30	
Artificial-g structure		2.30	
Resource Nodes (docking, prox ops)		3.00	
Airlock(s) (AL)		0.30	
Hyperbaric airlock(s) (HAL)		0.70	
Radiation shelter shielding		2.00	
Life support system (LSS)		1.40	
Data management system (DMS)		0.30	
Internal Com/EPS/TCS		1.80	
Spaceborne External Services			1.08
Electrical power system (EPS), external		0.28	
Thermal control system (TCS), external		0.30	
Communications system, external		0.50	

AOTPM

STVSIZER dated 2/9/88 revised 4/2/89

Mars Orbital Maneuvering Vehicle (MOMV) sized for
MCSV delivery from 250 circular to Phobos (no plane change).

02-Apr-89 Date of Run
08:19 PM Time of Run

STV design factors	
Tank % of prop	0.1
Aerobrake	0.15
Aerobrake cell	0
Isp	455
Structure	0.02
Min Stage T/W	0.75
Engine T/W	30

STV Weight Calculations	
Propellant	12647.89
Engine (2 RL-10B-2's)	382
Tank	1264.789
Aerobrake	0
Structure	304.6342
Dry weight	2583.818
Weight error (sum check)	9.1E-13
Gross weight w/o payload	15231.71
Flight Reserve Propellant	632.3946
Total Propellant Weight	13280.28
Thrust	11460
Mass fraction	0.830365

Mission segments	Delta-V	k value	(k-1)/k	g (mat
Phase one delta V	1.225	1.3161690	0.240219	0.0098
Phase two delta V	0	1	0	
Phase three delta V	1.225	1.3161690	0.240219	
Phase four delta V	0	1	0	
Phase five delta V	0	1	0	
Phase six delta V	0	1	0	
Phase seven delta V	0	1	0	
Phase eight delta V	0	1	0	
Phase nine delta V	0	1	0	
Phase ten delta V	0	1	0	

	Delta wt	Current wt	T/W
Pre-mission phase			
Initial payload loading	0		
Stage gross w/o p/l	15231.71		
Initial Weight		15231.712	0.75
Mission Phase one			
Mass off-load	0		
Propellant Used	3658.949		
Mission Phase two			
Current Weight		11572.762	0.99
Mass off-load	-25847		
Propellant Used	0		
Mission Phase three			
Current Weight		37419.762	0.31
Mass off-load	0		
Propellant Used	8988.944		
Mission Phase four			
Current Weight		28430.818	0.40
Mass off-load	25847		
Propellant Used	0		
Mission Phase five			
Current Weight		2583.8183	4.44
Mass off-load	0		
Propellant Used	0		
Mission Phase six			
Current Weight		2583.8183	4.44
Mass off-load	0		
Propellant Used	0		
Mission Phase seven			
Current Weight		2583.8183	4.44

Mass off-load	0		
Propellant Used	0		
Mission Phase eight			
Current Weight		2583.8183	4.44
Mass off-load	0		
Propellant Used	0		
Mission Phase nine			
Current Weight		2583.8183	4.44
Mass off-load	0		
Propellant Used	0		
Mission Phase ten			
Current Weight		2583.8183	4.44
Mass off-load	0		
Propellant Used	0		
Final weight		2583.8183	4.44
Total prop used	12647.89		

Reference Excursion Vehicles for Mars Evolution dated 4/2/89

Sizes Lander/Aerobrake/Propulsion Module (LAPM), Ascent Propellant Module (APM), and the Crew Cab

31-May-89 Date of Run
08:43 AM Time of Run

Constants					
Earth G	9.816				
lbsf to Newtons	4.452537				
lbsm to kg	0.4536				
Mars G	3.73008				
Lander/Aeroshell Module (LAM)					
Design Factors					
Delta-V	871.5	(m/s)			
Engine Thrust	66700	(N)	14980.2	(lbf)	RS-44
Engine Mass	155	(kg)	341.7	(lbm)	
Engine Isp	4545	(N/kg/s)	463.0	(sec)	
Isp margin (%)	1				
Working Isp	4499.55		458.4	(sec)	
Mixture Ratio (OX/Fuel)	6				
Number of Engines	6		T/W Mars	1.594699	
Fuel tank fraction	0.1				
Ox tank fraction	0.02				
Bulk margin (%)	1				
RCS prop frac of gross	1				
Legs (% supported mass)	3				
Structure (% dry)	15				
Aerobrake Core fraction	0.05				
Aerobrake Skirt fract	0.04				
Dry Mass					
Avionics (computers)	100	(kg)			
RCS System	67.27909	(kg)			
Fuel Tank	170.9314	(kg)			
Oxidizer Tank	205.1177	(kg)			
Refrigeration System	0	(kg)			
Landing Legs	1662.970	(kg)			
Total Engine Mass	930	(kg)			
Aerobrake Core	3363.954	(kg)			
Aerobrake Skirt	2691.163	(kg)			
Structure	1622.014	(kg)			
Total Dry Mass	10813.43	(kg)			
Inert Mass					
RCS Propellant	672.7909	(kg)			
Residual Fuel	16.92390	(kg)			
Residual Oxidizer	101.5434	(kg)			
Ascent Sys (optional)	0	(kg)			
APM + Up Payload (Opt)	0	(kg)	(D97)		
Down Payload	43827.67	(kg)			
Total Inert Mass	55432.36	(kg)	(Includes Payload)		
Inert without Payload	11604.69	(kg)			

```
 Wet Mass
   Usable Fuel Mass         1692.390 (kg)
   Usable Oxidizer Mass     10154.34 (kg)
 Total Wet Mass (Gross)     67279.09 (kg)     (Includes Payload)
 Gross without Payload      23451.42 (kg)

 Sizing Process
   1=reset, 0=run                  0
   Gain                           10
   Propellant Mass          11846.73 (kg)
   Actual Delta-V           871.5000 (m/s)
   Delta-V Error            0.000000 (m/s)


Ascent Propellant Module (APM)
 Design Factors
   Delta-V                      4100
   Engine Thrust               66700 Same Engines as LAM
   Engine Mass                   155 (Don't Change values here)
   Engine Isp                   4545 .
   Isp margin (%)                  1 .
   Working Isp               4499.55 .
   Mixture Ratio (OX/Fuel)         6 .
   Number of Engines               6 .      T/W Mars 1.764247
   Fuel tank fraction           0.15 400 Day storage
   Ox tank fraction             0.05
   Bulk margin (%)                 1
   Structure (% dry)              10

 Dry Mass
   Fuel Tank                787.0200 (kg)
   Oxidizer Tank            1574.040 (kg)
   Structure                279.0066 (kg)
   Propellant Feed System        150 (kg)
 Dry Mass Total             2790.066 (kg)

 Inert Mass
   Residual Fuel            51.94852 (kg)
   Residual Oxidizer        311.6911 (kg)
   Up Payload               10482.32 (kg)     (Hook to other stages)
 Inert Mass Total           13636.03 (kg)     (Includes Payload)
 Inert without payload      3153.706 (kg)

 Wet Mass
   Usable Fuel              5194.852 (kg)
   Usable Oxidizer          31169.11 (kg)
 Wet Mass Total             49999.99 (kg)     (Includes Payload)
 Gross without payload      39517.67 (kg)

 Combined LAM/APM Masses
   Inert Mass               24449.46 (kg)     (APM Inert + LAM Dry)
   Wet Mass (Gross)         60813.43 (kg)     (APM Wet + LAM Dry)


 Sizing Process
   1=reset, 0=run                  0
```

Gain	1	
Propellant Mass	36363.96	(kg)
Actual Delta-V	4099.999	(m/s)
Delta-V Error	-0.00002	(m/s)

Crew Cab Module

Design Factors

Design Crew Count	7	
Actual Crew Count	7	
Crew, Suit, Consum	1470	(kg)
Return Cargo	200	(kg)
Cab Mass	2640	(kg)
Total Cab Mass	4310	(kg)

NTR Vehicle Design Point

Engine	10.0 t		Produces 5000 MWth, 900 sec Isp
Shield	10.0		
Propellant	225.0		Hydrogen
Tankage	25.0		
Aeroshell	25.8		
Payload	60.0	-----	High Energy Orbit, Round Trip
Payload	-----	185.0 t	Minimum Energy Orbit, Round Trip
Total	355.8 t	480.8 t	

Minimum Energy Transfer Time	220-300 days
High Energy Transfer Time	100-170 days

Artificial Gravity of 0.6 g can be provided by rotating the ship at 4 rpm.

If return propellant is available at Phobos the above performance can be improved, allowing the ship to carry 310 t payloads on minimun energy missions or 60 t payloads on high energy missions with flight times of 80-130 days.

NEP Cargo Vehicle Design Point

Reactor	13.7 t	Produces 26.7 MWth
Shield	3.3 t	65 cm thick LiH/W
Turbines	20.8 t	Closed loop Brayton cycle generators producing 5 MWe
Radiator	20.3 t	3694 square meters
Propulsion	23.5 t	Ion Thrusters and PPU, 3.65 MW, 6000 sec Isp
Structure	10.0 t	Single truss
Propellant	167.6 t	Argon
Tankage	16.8 t	
Payload	400.0 t	One way. Returns to Earth empty.
Total	676.0 t	

Earth Escape Burn Time	387 days
Mid-Course Burn Time	298 days
Mid-Course Coast Time	2 days
Phobos Capture Burn Time	96 days
Total Trans-Mars Mission Time	783 days

Mission Summary Sheet

Mission: BCC-EK
L02.Hum3c.ChHO.MAb.1D3cP.19s0uR.ChHO.ECCVd
L01.Car.ChHO.MAb

Reference mission: BCC-EK *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Human traj: Exped. baseline traj: 2002 sprint nominal
Cargo traj: Exped. baseline traj: 2001 nominal cargo mission
Override(s) in effect: time(s) only.

Mission purpose:
TIC-1R. Final CS-2.1 Mars Exped. TEIS on MPV, time o'rides, actual ΔV's
hi boil; 2 act MAb's, 11 t 1-stg MAV, Fixed TMI & TEI tanks, 10 t down

Total IMLEO = 824.90 t

		Human mission mass (t)	Cargo mission mass (t)
M(C1)	Beginning of phase C (Earth escape, TMI)	615.93	208.97
M(D1)	Beginning of phase D (transfer to Mars)	197.34	77.62
M(D2)	After MCC, DSM, and dumps	186.06	74.75
M(E1)	Beginning of phase E (MOC)	186.06	74.75
M(Ef)	After achieving capture (& burns and/or drops)	151.90	42.50
M(F1)	Beginning of Mars orbital operations	151.90	42.50
M(Ff)	End of Mars orbital operations	148.02	1.51
M(G1)	Beginning of phase G (Mars escape)	148.02	
M(H1)	Beginning of phase H (transfer to Earth)	39.48	
M(H2)	After MCC and DSM	37.58	
M(I1)	ECCV at beginning of EEL	5.64	
M(I2)	ECCV after EEL	5.64	
TMI propellant		380.79	118.75
TEI propellant		92.21	
all other propellant		14.31	12.16
H/O propellant reserve		18.07	5.25

Launch dates: Cargo: 04/15/2001. Human: 09/03/2002. Hum Roundtrip: 740 d
Technology status: Nominal
Stg/rec/Ab: TMIS-3, TEIS-1, MAb. Recover: ECCVd (16.0 km/s)
No artificial gravity
IMM structure: cyl. mods: none; disk mods: 2 @ 25.0 ft; tunnel sect: 2
EPS: Spaceborne: 11 kW solar
Landed: 2 kW solar 1 kW RTG 30 kW-hr NiH2
Propellant reserve: margin comb: sum; H/O: 23.32 t; Biprop: 0.63 t; other: -0.00 t
Tankage factors: TMIS: 7%; TEIS: 15%
Boiloff: H/O: 59.62 t; other: 0.00 t
Science equip: ISE: 0.85 t; MOSE: 0.30 t; MLSE: 3.26 t; RVR: 0.00 t
Consumables: food: 3.94 t; water: 0.02 t; other: 5.51 t

Human Mission Mass Allocation Report

Mission: BCC-EK
L02.Hum3c.ChHO.MAb.1D3cP.19s0uR.ChHO.ECCVd
L01.Car.ChHO.MAb

Reference mission: BCC-EK *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Human traj: Exped. baseline traj: 2002 sprint nominal
Cargo traj: Exped. baseline traj: 2001 nominal cargo mission
Override(s) in effect: time(s) only.

Mission purpose:
TIC-1R. Final CS-2.1 Mars Exped. TEIS on MPV, time o'rides, actual ΔV's
hi boil; 2 act MAb's, 11 t 1-stg MAV, Fixed TMI & TEI tanks, 10 t down

	Mass (t)			
MSS (IMLEO)				615.93
TMIS (Trans-Mars Injection System)			418.59	
Stage 1		139.39		
Propellant (LH2/LOX)	126.79			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
Stage 2		139.60		
Propellant (LH2/LOX)	127.00			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
Stage 3		139.60		
Propellant (LH2/LOX)	127.00			
Tank(s)	8.89			
Engine(s), avionics+	3.71			
MTV			197.34	

MTV (Mars Transfer Vehicle)				197.34
Crew consumables, Mars transfer phase			4.10	
Artificial-g (2 spin-up/downs)			0.00	
Propellant (Stored biprop)		0.00		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
DSM (ETM) (deep space maneuver, Earth-to-Mars)			0.00	
Propellant (LH2/LOX)		0.00		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
MCC (ETM) (mid-course correction)			3.86	
Propellant (Stored biprop)		3.20		
Tank(s)		0.16		
Engine(s), avionics+		0.50		
RCS (ETM) (reaction control system)			3.32	
Propellant (Stored biprop)		3.32		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
Venus swingby probes(s)			0.00	
MOCS (Mars orbital capture system)			34.16	
Mars capture Ab		31.00		
Propulsion		3.16		
Propellant (LH2/LOX)	3.16			
Tank(s)	0.00			

Engine(s), avionics+	0.00			
MOV (F1)			151.90	
MOV (Mars Orbiting Vehicle; Phase F1--just after MOC)				151.90
Crew consumables, Mars orbit			0.12	
MOO 1 (Mars orbital operations -1)			0.78	
Propellant (LH2/LOX)		0.78		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
MOO 2			0.00	
Propellant (LH2/LOX)		0.00		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
RCS (MOO)			2.92	
Propellant (Stored biprop)		2.55		
Tank(s)		0.29		
Engine(s), avionics+		0.07		
Satellites			0.00	
RelayComSat(s)		0.00		
MarsSciSat(s)		0.00		
Ph/D teleoperator(s)		0.00		
Teleoperated MRSR			0.00	
MOSE (Mars orbital science equipment)			0.15	
Earth returnables (received)			-0.10	
MOV (Ff)			148.02	
MOV (Mars Orbiting Vehicle; Phase Ff--just prior to TEI)				148.02
TEIS (trans-Earth injection system)			108.54	
Propellant (LH2/LOX)		92.21		
Tank(s)		14.55		
Engine(s), avionics+		1.78		
ETV			39.48	
ETV (Earth Transfer Vehicle)				39.48
IMM (interplanetary mission module)			25.45	
Spaceborne external services (power, com, thermal)			0.97	
Crew consumables, transfer to Earth			4.72	
Artificial-g (2 spin-up/downs)			0.00	
Propellant (Stored biprop)		0.00		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
Flyaround			0.00	
Propellant (LH2/LOX)		0.00		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
DSM (MTE)			0.00	
Propellant (Stored biprop)		0.00		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
MCC (MTE)			1.12	
Propellant (Stored biprop)		0.64		
Tank(s)		0.03		
Engine(s), avionics+		0.45		
RCS (MTE)			0.78	
Propellant (Stored biprop)		0.66		
Tank(s)		0.03		
Engine(s), avionics+		0.08		
Spacesuits			0.35	
ISE (interplanetary science equipment)			0.45	
Solar/SPE monitoring		0.20		

Astro/Planetary		0.25		
ECCV			5.64	
ECCV (Earth Crew Capture Vehicle)				5.64
Payload			4.90	
Crew+returnables+consum+suits		0.48		
Inert module		4.42		
EELS (Earth entry & landing system)			0.74	
Earth entry Ab		0.49		
Propulsion		0.00		
Propellant (Stored biprop)	0.00			
Tank(s)	0.00			
Engine(s), avionics+	0.00			
Other EELS (parachutes, avionics)		0.25		
IMM (Interplanetary Mission Modules)				25.45
Cylindrical Module(s)			0.00	
Disk Module(s)			18.00	
Tunnel(s)			0.60	
Tanks for crew consumables			0.45	
Resource Nodes (docking, prox ops)			0.00	
Airlock(s) (AL)			0.30	
Hyperbaric airlock(s) (HAL)			0.00	
Radiation shelter shielding			1.00	
Life support system (LSS)			2.00	
Data management system (DMS)			0.60	
Internal Com/EPS/TCS			2.50	
Spaceborne External Services				0.97
Electrical power system (EPS), external			0.22	
Thermal control system (TCS), external			0.40	
Communications system, external			0.35	

Human Mission
Crew Consumables* Report

Mission: BCC-EK
L02.Hum3c.ChHO.MAb.1D3cP.19s0uR.ChHO.ECCVd
L01.Car.ChHO.MAb

Reference mission: BCC-EK *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Human traj: Exped. baseline traj: 2002 sprint nominal
Cargo traj: Exped. baseline traj: 2001 nominal cargo mission
Override(s) in effect: time(s) only.

Mission purpose:
TIC-1R. Final CS-2.1 Mars Exped. TEIS on MPV, time o'rides, actual ΔV's
hi boil; 2 act MAb's, 11 t 1-stg MAV, Fixed TMI & TEI tanks, 10 t down

Crew composition: Nominal U. S. male crew

Period	Mission phase	# of crew	Time	Person-days	Margin	Total mass (t)
LEO Checkout	A	3	21 day	63	15 %	0.26
MTV	D	3	330 day	990	15 %	4.10
MOV	F		30 day	30	15 %	0.12
MDV	F3	3	19 sol	60	5 %	0.22
ETV	H	3	380 day	1140	15 %	4.72
ECCV	I	3	1 day	3	200 a %	0.05
					Total (incl. margin)	9.47
					Total (w/o margin)	8.23

Total supply = 6.26 person-years = 2286 person-days
Average supply = 4.14 kg/person-day

Water: crew prod. = 3.4 kg/p-d; hygiene = 8.0 kg/p-d; recycling efficiency = 90.0 %

Reference Consumables Rate for Nominal U. S. Male Crew (kg/person-day):

	Food	Pot. Water (Recycled)	Other	Total
Spaceborne	1.5	1.0	2.1	4.6
Surface	1.5	2.0	2.0	5.5
MAV, ECCV	1.5	2.0	2.0	5.5
Mission Totals (t) (includes LEO checkout)				
Consumption	3.43	0.01	4.79	8.23
Initial Storage	3.94	0.02	5.51	9.47

* Consumables includes LSS + Food
(a) To provide interplanetary safe-haven capability.

Human Mission
Propulsion System Summary Report

Mission: BCC-EK
L02.Hum3c.ChHO.MAb.1D3cP.19s0uR.ChHO.ECCVd
L01.Car.ChHO.MAb

Reference mission: BCC-EK *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Human traj: Exped. baseline traj: 2002 sprint nominal
Cargo traj: Exped. baseline traj: 2001 nominal cargo mission
Override(s) in effect: time(s) only.

Mission purpose:
TIC-1R. Final CS-2.1 Mars Exped. TEIS on MPV, time o'rides, actual ΔV's
hi boil; 2 act MAb's, 11 t 1-stg MAV, Fixed TMI & TEI tanks, 10 t down

Element	Delta V (km/s)	Prop. Sys. #	Propellant	Isp (lbf-s/lbm)	Mi/Mf	Total thrust (lbf)	Acceleration (gee) Initial	Acceleration (gee) Final	Burn time (min)
TMI stage 1	0.949	1	LH2/LOX	471	1.228	532000	0.40	0.49	3.7
TMI stage 2	1.279	2	LH2/LOX	471	1.319	532000	0.52	0.68	3.7
TMI stage 3	1.962	3	LH2/LOX	471	1.529	532000	0.74	1.13	3.7
ETM DSM	0.000	11	LH2/LOX	470	1.000	120000	0.00	0.00	0.0
ETM MCC	0.050	9	Stored biprop	316	1.016	18007	0.04	0.04	2.0
RCS ETM	0.050	17	Stored biprop	310	1.017	4000	0.01	0.01	9.2
MOC post A/C peri. raise	0.066	11	LH2/LOX	470	1.014	120000	0.29	0.30	0.4
MOO 1	0.020	11	LH2/LOX	470	1.004	120000	0.37	0.37	0.1
MOO 2	0.000	11	LH2/LOX	470	1.000	120000	0.00	0.00	0.0
RCS MOO	0.050	17	Stored biprop	310	1.017	4000	0.01	0.01	7.1
TEI	3.847	11	LH2/LOX	470	2.304	120000	0.38	0.89	11.5
MTE DSM	0.000	13	Stored biprop	316	1.000	18007	0.00	0.00	0.0
MTE MCC	0.050	13	Stored biprop	316	1.016	18007	0.21	0.21	0.4
RCS MTE	0.050	18	Stored biprop	311	1.017	900	0.01	0.01	8.2
ECCV decel. before DE	0.000	34	Stored biprop	340	1.000	3754	0.00	0.00	0.0

◊ Indicates that value overrides that in engines/tanks data base.

Human Mission
Propulsion Engines Report

Mission: BCC-EK
L02.Hum3c.ChHO.MAb.1D3cP.19s0uR.ChHO.ECCVd
L01.Car.ChHO.MAb

Reference mission: BCC-EK *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Human traj: Exped. baseline traj: 2002 sprint nominal
Cargo traj: Exped. baseline traj: 2001 nominal cargo mission
Override(s) in effect: time(s) only.

Mission purpose:
TIC-1R. Final CS-2.1 Mars Exped. TEIS on MPV, time o'rides, actual ΔV's
hi boil; 2 act MAb's, 11 t 1-stg MAV, Fixed TMI & TEI tanks, 10 t down

Element	Propellant	Engine Ident.	Engine name	Rev.	Isp (lbf-s/lbm)	Mass per engine (kg)	Thrust per engine (lbf)	# of engines	Avionics+ (kg)
TMI stage 1	LH2/LOX	14	SSME, adv, 306:1	2	471	3560.2	532000	1	150.0
TMI stage 2	LH2/LOX	14	SSME, adv, 306:1	2	471	3560.2	532000	1	150.0
TMI stage 3	LH2/LOX	14	SSME, adv, 306:1	2	471	3560.2	532000	1	150.0
ETM DSM	LH2/LOX	4	RL-10-X1 (paper e	5	470	272.2	20000	6	150.0
ETM MCC	Stored bipro	33	Shuttle OMS	2	316	134.0	6002	3	100.0
RCS ETM	Stored bipro	48	Marquardt R-40B (	2	310	13.6	1000	4	20.0
MOC post A/C peri. raise	LH2/LOX	4	RL-10-X1 (paper e	5	470	272.2	20000	6	150.0
MOO 1	LH2/LOX	4	RL-10-X1 (paper e	5	470	272.2	20000	6	150.0
MOO 2	LH2/LOX	4	RL-10-X1 (paper e	5	470	272.2	20000	6	150.0
RCS MOO	Stored bipro	48	Marquardt R-40B (	2	310	13.6	1000	4	20.0
TEI	LH2/LOX	4	RL-10-X1 (paper e	5	470	272.2	20000	6	150.0
MTE DSM	Stored bipro	33	Shuttle OMS	2	316	134.0	6002	3	50.0
MTE MCC	Stored bipro	33	Shuttle OMS	2	316	134.0	6002	3	50.0
RCS MTE	Stored bipro	47	Marquardt R-4D	2	311	3.8	100	9	50.0
ECCV decel. before DE	Stored bipro	42	XLR - 132	2	340	57.0	3754	1	25.0

◊ Indicates that value overrides that in engines/tanks data base.

Human Mission
Propulsion Tanks Report

Mission: BCC-EK
L02.Hum3c.ChHO.MAb.1D3cP.19s0uR.ChHO.ECCVd
L01.Car.ChHO.MAb

Reference mission: BCC-EK *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Human traj: Exped. baseline traj: 2002 sprint nominal
Cargo traj: Exped. baseline traj: 2001 nominal cargo mission
Override(s) in effect: time(s) only.

Mission purpose:
TIC-1R. Final CS-2.1 Mars Exped. TEIS on MPV, time o'rides, actual ΔV's
hi boil; 2 act MAb's, 11 t 1-stg MAV, Fixed TMI & TEI tanks, 10 t down

Element	Propellant	Tank Ident.	Tank name	Rev.	Tankage Fact.(%)	Tankage Cap.(t)	Margins(%)* Bulk	Margins(%)* dV	Margins(%)* Isp	Boiloff(%/mo.) LEO	Boiloff(%/mo.) IP(1AU)**	Boiloff(%/mo.) MO	Post dV Dispos.
TMI stage 1	LH2/LOX	9	TMI,med-tf,med-bo	4	7.0	127.0	1.0	1.0	1.0	3.000	1.000	0.500	Drop
TMI stage 2	LH2/LOX	9	TMI,med-tf,med-bo	4	7.0	127.0	1.0	1.0	1.0	3.000	1.000	0.500	Drop
TMI stage 3	LH2/LOX	9	TMI,med-tf,med-bo	4	7.0	127.0	1.0	1.0	1.0	3.000	1.000	0.500	Drop
ETM DSM	LH2/LOX	15	CS-2.1 Fixed TEI	2	15.0	97.0	1.0	1.0	1.0	0.550	1.000	0.330	Save
ETM MCC	Stored bipro	46	Small biprop (MCC	2	5.0	Rubber	1.0	1.0	1.0	0.000	0.000	0.000	Drop
RCS ETM	Stored bipro	46	Small biprop (MCC	2	5.0	Rubber	1.0	1.0	1.0	0.000	0.000	0.000	Save
MOC post A/C peri. raise	LH2/LOX	15	CS-2.1 Fixed TEI	2	15.0	97.0	1.0	1.0	1.0	0.550	1.000	0.330	Save
MOO 1	LH2/LOX	15	CS-2.1 Fixed TEI	2	15.0	97.0	1.0	1.0	1.0	0.550	1.000	0.330	Save
MOO 2	LH2/LOX	15	CS-2.1 Fixed TEI	2	15.0	97.0	1.0	1.0	1.0	0.550	1.000	0.330	Save
RCS MOO	Stored bipro	46	Small biprop (MCC	2	5.0	Rubber	1.0	1.0	1.0	0.000	0.000	0.000	Drop
TEI	LH2/LOX	15	CS-2.1 Fixed TEI	2	15.0	97.0	2.0	1.0	1.0	0.550	1.000	0.330	Drop
MTE DSM	Stored bipro	46	Small biprop (MCC	2	5.0	Rubber	1.0	1.0	1.0	0.000	0.000	0.000	Save
MTE MCC	Stored bipro	46	Small biprop (MCC	2	5.0	Rubber	1.0	1.0	1.0	0.000	0.000	0.000	Drop
RCS MTE	Stored bipro	46	Small biprop (MCC	2	5.0	Rubber	1.0	1.0	1.0	0.000	0.000	0.000	Drop
ECCV decel. before DE	Stored bipro	41	MAV biprop	3	3.0	Rubber	1.0	1.0	1.0	0.000	0.000	0.000	Drop

* Combination of reserves due to margins: sum
** Interplanetary trajectory boiloff thermal factor (relative to 1 A. U.):
Cargo = 70.0 % Human outbound = 120.0 % Human inbound = 100.0 %

◊ Indicates that value overrides that in engines/tanks data base.

Human Mission
Total Propellant* Report

Mission: BCC-EK
L02.Hum3c.ChHO.MAb.1D3cP.19s0uR.ChHO.ECCVd
L01.Car.ChHO.MAb

Reference mission: BCC-EK *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Human traj: Exped. baseline traj: 2002 sprint nominal
Cargo traj: Exped. baseline traj: 2001 nominal cargo mission
Override(s) in effect: time(s) only.

Mission purpose:
TIC-1R. Final CS-2.1 Mars Exped. TEIS on MPV, time o'rides, actual ΔV's
hi boil; 2 act MAb's, 11 t 1-stg MAV, Fixed TMI & TEI tanks, 10 t down

		Propellant Mass Expended (t)			
		Main		ACS/RCS	
Element	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI stage 1	C	126.792	0.000	0.000	0.000
TMI stage 2	C	127.000	0.000	0.000	0.000
TMI stage 3	C	127.000	0.000	0.000	0.000
Mars transfer (DSM)	D	0.000	0.000	0.000	0.000
Mars transfer (Art-g)	D	0.000	0.000	0.000	0.000
Mars transfer (MCC, RCS)	D	0.000	3.198	0.000	3.315
MOC	E	3.165	0.000	0.000	0.000
Mars orbit (RCS)	F	0.000	0.000	0.000	2.552
Mars orbit (MOO 1)	F	0.781	0.000	0.000	0.000
Mars orbit (MOO 2)	F	0.000	0.000	0.000	0.000
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
TEI	G	92.209	0.000	0.000	0.000
Flyaround	G	0.000	0.000	0.000	0.000
Earth transfer (DSM)	H	0.000	0.000	0.000	0.000
Earth transfer (Art-g)	H	0.000	0.000	0.000	0.000
Earth transfer (MCC, RCS)	H	0.000	0.638	0.000	0.661
ECCV (EELS)	I	0.000	0.000	0.000	0.000
	Totals	476.946	3.836	0.000	6.529

*Includes boiloff and reserves for Isp, delta V, and bulk propellant margins.

Human Mission Propellant Reserves* Report

Mission: BCC-EK
L02.Hum3c.ChHO.MAb.1D3cP.19s0uR.ChHO.ECCVd
L01.Car.ChHO.MAb

Reference mission: BCC-EK *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Human traj: Exped. baseline traj: 2002 sprint nominal
Cargo traj: Exped. baseline traj: 2001 nominal cargo mission
Override(s) in effect: time(s) only.

Mission purpose:
TIC-1R. Final CS-2.1 Mars Exped. TEIS on MPV, time o'rides, actual ΔV's
hi boil; 2 act MAb's, 11 t 1-stg MAV, Fixed TMI & TEI tanks, 10 t down

		Propellant Reserve (t)			
		Main		ACS/RCS	
Element	Mission phase	Cryo	Biprop	Cryo	Biprop
TMI stage 1	C	3.855	0.000	0.000	0.000
TMI stage 2	C	4.046	0.000	0.000	0.000
TMI stage 3	C	4.456	0.000	0.000	0.000
Mars transfer (DSM)	D	0.000	0.000	0.000	0.000
Mars transfer (Art-g)	D	0.000	0.000	0.000	0.000
Mars transfer (MCC, RCS)	D	0.000	0.095	0.000	0.098
MOC	E	0.080	0.000	0.000	0.000
Mars orbit (RCS)	F	0.000	0.000	0.000	0.075
Mars orbit (MOO 1)	F	0.019	0.000	0.000	0.000
Mars orbit (MOO 2)	F	0.000	0.000	0.000	0.000
RelayComSat(s)	F	0.000	0.000	0.000	0.000
MarsSciSat(s)	F	0.000	0.000	0.000	0.000
Ph/D teleoperator(s)	F	0.000	0.000	0.000	0.000
Teleoperated MRSR(s)	F	0.000	0.000	0.000	0.000
TEI	G	5.610	0.000	0.000	0.000
Flyaround	G	0.000	0.000	0.000	0.000
Earth transfer (DSM)	H	0.000	0.000	0.000	0.000
Earth transfer (Art-g)	H	0.000	0.000	0.000	0.000
Earth transfer (MCC, RCS)	H	0.000	0.019	0.000	0.020
ECCV (EELS)	I	0.000	0.000	0.000	0.000
	Totals	18.066	0.113	0.000	0.193

*Reserves for Isp, delta V, and bulk propellant margins.

Elements Report
Human Mission

	-----Dry Mass (t)-------		-----Gross Mass (t)-----	
Manned vehicles (w/o consumables)				
ITV (w/o science and satellites)		59.05		69.42
IMM	25.45		25.45	
Ab (MOCS)	31.00		31.00	
Interstructure	0.00		0.00	
External services	0.97		0.97	
RCS	0.48		7.01	
MCC	1.15		4.98	
ECCV		5.16		5.25
Propulsion systems				
TMI stage 1		12.60		139.39
TMI stage 2		12.60		139.60
TMI stage 3		12.60		139.60
MOC/MOO 1/TEI stage 1		16.33		112.49
Science				
ISE		0.45		0.45
Venus probes		0.00		0.00
MOSE		0.15		0.15
Mars satellites		0.00		0.00
MRSR		0.00		0.00
Other				
Crew		0.00		0.24
Consumables		0.00		8.99
Spacesuits		0.00		0.35
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Totals		118.95		615.93

Human Mission Astrodynamics Report

Mission: BCC-EK
L02.Hum3c.ChHO.MAb.1D3cP.19s0uR.ChHO.ECCVd
L01.Car.ChHO.MAb

Reference mission: BCC-EK *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Human traj: Exped. baseline traj: 2002 sprint nominal
Cargo traj: Exped. baseline traj: 2001 nominal cargo mission
Override(s) in effect: time(s) only.

Mission purpose:
TIC-1R. Final CS-2.1 Mars Exped. TEIS on MPV, time o'rides, actual ΔV's
hi boil; 2 act MAb's, 11 t 1-stg MAV, Fixed TMI & TEI tanks, 10 t down

Trajectory Type: Exped. baseline traj: 2002 sprint nominal

	Date	Apoapsis (km)	Periapsis (km)	Inclination (deg)	Entry Vel. (km/s)	C3 (km^2/s^2)
Earth departure (TMI)	09/03/2002	500	500	28.50		23.35
Venus swingby	11/12/2002					
Mars arrival (MOC)	06/15/2003	300	300	76.35	8.035	40.13
Mars departure (TEI)	07/15/2003	300	300	76.35		29.41
Earth arrival (EOC)	01/01/2004	500	500	28.50	12.069	22.76

Duration

	Trajectory			Override		
	Sols	Days	Months	Days	Months	
Ave. pre-TMI time in LEO						
TMIS		91.3	3.00			
MTV		91.3	3.00			
Marsbound (ETM)		285.8	9.39	330.0	10.84	Use override time
Mars orbit	29.2	30.0	0.99	0.0	0.00	Use traj. time
Mars surface						
MDV	19.4					
Earthbound (MTE)		169.2	5.56	380.0	12.48	Use override time
Total trip		485.0	15.93	710.0	23.33	

Delta V Summary

Item	Delta V (km/s) Trajectory	Override	
TMI	4.190	4.410	Use trajectory delta V
ETM DSM	0.000	0.000	Use trajectory delta V
ETM MCC	0.050		
RCS ETM	0.050		
MOC post A/C periapsis raise	0.066	0.100	Use trajectory delta V
MOO 1	0.020		
MOO 2	0.000		
RCS MOO	0.050		
MDV deorbit	0.025		
MDV terminal descent	0.585		

MDV hover for 30.0 s	0.112		
TEI	3.847	3.900	Use trajectory delta V
MTE DSM	0.000	0.000	Use trajectory delta V
MTE MCC	0.050		
RCS MTE	0.050		
ECCV decel. before DE	0.000		

Mars Descent and Ascent Vehicles Mass Allocation Report

Mission: BCC-EK
L02.Hum3c.ChHO.MAb.1D3cP.19s0uR.ChHO.ECCVd
L01.Car.ChHO.MAb

Reference mission: BCC-EK *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Human traj: Exped. baseline traj: 2002 sprint nominal
Cargo traj: Exped. baseline traj: 2001 nominal cargo mission
Override(s) in effect: time(s) only.

Mission purpose:
TIC-1R. Final CS-2.1 Mars Exped. TEIS on MPV, time o'rides, actual ΔV's
hi boil; 2 act MAb's, 11 t 1-stg MAV, Fixed TMI & TEI tanks, 10 t down

	Mass (t)			
MDV (w/o crew)				35.20
MELS (Mars entry and landing system)			12.07	
Mars entry Ab (5.0 %)		1.67		
Deorbit propulsion		0.29		
Propellant (Stored biprop)	0.29			
Tank(s)	0.00			
Engine(s), avionics+	0.00			
Parachute		2.71		
All-propulsive descent option		0.00		
Propellant (Stored biprop)	0.00			
Tank(s)	0.00			
Engine(s), avionics+	0.00			
Terminal descent propulsion		7.40		
Propellant (Stored biprop)	6.37			
Tank(s)	0.20			
Engine(s), avionics+	0.83			
Adapter structure (incl. landing legs)			2.12	
Landed P/L (w/o crew)			10.00	
MLMM		6.15		
MLOE		3.44		
Crew*		(0.24)		
Mars landed expendables (MLX)		0.42		
Crew consumables	0.22			
other	0.20			
MAV (w/o crew)			11.00	
MLOE (Mars Landed Operation Equipment)				3.44
MLSE (Mars landed science equipment)			3.26	
MLSE inside MLMM		0.25		
MLSE outside MLMM		3.01		
MLTE (Mars landed transportation equipment)			0.18	
Manned rover(s)		0.00		
Surface suits		0.18		
Teleoperated equip.			0.00	
MLCE (Mars landed construction equipment)			0.00	
MLME (ISRU demo's) (Mars landed mfg. eq.; in situ resource util.)			0.00	
MLMM (Mars Landed Mission Modules)				6.15
Structure			2.30	
Pressure shell, support structure		2.30		

Cylin. modules, SS, fully outfitted	0.00		
Cylin. modules, SS, partially outfitted	0.00		
Cylin. modules, SS, shells	0.00		
Disk modules, 22 ft. diameter	2.00		
Disk modules, 25 ft. diameter	0.00		
Disk modules, 31 ft. diameter	0.00		
Tunnels (3-m sections)	0.30		
Partitions, equipment racks		0.00	
Windows		0.00	
Tanks for crew consumables			0.01
Airlock (AL)			0.30
Hyperbaric airlock (HAL)			0.00
Man-systems			0.45
Living quarters		0.22	
Galley		0.12	
Personal hygiene		0.10	
ECLSS (environmental control & life support system)			1.10
DMS (data management system)			0.20
EPS (electrical power system)			0.99
Power sources		0.24	
Solar (PVPA)	0.15		
Nuclear reactor	0.00		
RTG	0.09		
Energy storage		0.75	
Ni/H2 batteries	0.75		
HEDRB batteries	0.00		
Regen. FC	0.00		
TCS (thermal control system)			0.45
Communications			0.35

* Used only to size propulsion. Crew+returnables allocated in ECCV.

Technology Status

MDV adapter scaling	Nominal

Cargo Mission
Mass Allocation Report

Mission: BCC-EK
L02.Hum3c.ChHO.MAb.1D3cP.19s0uR.ChHO.ECCVd
L01.Car.ChHO.MAb

Reference mission: BCC-EK *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Human traj: Exped. baseline traj: 2002 sprint nominal
Cargo traj: Exped. baseline traj: 2001 nominal cargo mission
Override(s) in effect: time(s) only.

Mission purpose:
TIC-1R. Final CS-2.1 Mars Exped. TEIS on MPV, time o'rides, actual ΔV's
hi boil; 2 act MAb's, 11 t 1-stg MAV, Fixed TMI & TEI tanks, 10 t down

	Mass (t)			
MCV (IMLEO)				208.97
TMIS			131.35	
Propellant (LH2/LOX)		118.75		
Tank(s)		8.89		
Engine(s), avionics+		3.71		
MTV			77.62	
MTV				77.62
MCC			1.57	
Propellant (Stored biprop)		1.30		
Tank(s)		0.04		
Engine(s), avionics+		0.23		
RCS (ETM)			1.30	
Propellant (Stored biprop)		1.30		
Tank(s)		0.00		
Engine(s), avionics+		0.00		
MOCS			32.25	
Mars capture Ab		31.00		
Propulsion		1.25		
Propellant (LH2/LOX)	1.25			
Tank(s)	0.00			
Engine(s), avionics+	0.00			
MOV (F1)			42.50	
MOV (F1)				42.50
MOO			1.94	
Propellant (LH2/LOX)		0.98		
Tank(s)		0.34		
Engine(s), avionics+		0.62		
RCS (MOO)			0.85	
Propellant (Stored biprop)		0.66		
Tank(s)		0.10		
Engine(s), avionics+		0.10		
Payload			38.75	
Satellites		3.00		
RelayComSat(s)	2.00			
MarsSciSat(s)	1.00			
Ph/D teleoperator(s)	0.00			
Teleoperated MRSR		0.00		
MOSE		0.15		

ISE		0.40	
Solar/SPE monitoring	0.25		
Astro/Planetary	0.15		
MDV(s)		35.20	
Structure			0.50
Support Services			0.46
Data management system (DMS)		0.05	
Electrical power system (EPS)		0.18	
Thermal control system (TCS)		0.15	
Communications system		0.07	

Cargo Mission
Propulsion System Summary Report

Mission: BCC-EK
L02.Hum3c.ChHO.MAb.1D3cP.19s0uR.ChHO.ECCVd
L01.Car.ChHO.MAb

Reference mission: BCC-EK *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Human traj: Exped. baseline traj: 2002 sprint nominal
Cargo traj: Exped. baseline traj: 2001 nominal cargo mission
Override(s) in effect: time(s) only.

Mission purpose:
TIC-1R. Final CS-2.1 Mars Exped. TEIS on MPV, time o'rides, actual ΔV's
hi boil; 2 act MAb's, 11 t 1-stg MAV, Fixed TMI & TEI tanks, 10 t down

Element	Delta V (km/s)	Prop. Sys. #	Propellant	Isp (lbf-s/lbm)	Mi/Mf	Total thrust (lbf)	Acceleration (gee) Initial	Acceleration (gee) Final	Burn time (min)
TMI	3.514	36	LH2/LOX	471	2.140	532000	1.18	2.53	3.5
ETM MCC	0.050	40	Stored biprop	305	1.017	9892	0.06	0.06	1.4
RCS ETM	0.050	43	Stored biprop	311	1.017	1200	0.01	0.01	12.0
MOC post A/C peri. raise	0.066	41	LH2/LOX	460	1.015	66000	0.40	0.41	0.3
MOO	0.100	41	LH2/LOX	460	1.022	66000	0.78	0.79	0.2
RCS MOO	0.050	43	Stored biprop	311	1.017	1200	0.01	0.01	6.1
MDV deorbit	0.025	22	Stored biprop	316	1.008	36014	0.46	0.46	0.1
MDV terminal descent	0.585	22	Stored biprop	316	1.208	36014	0.55	0.66	1.6
MDV hover, 30.0 s	0.112	22	Stored biprop	316	1.037	36014	0.64	0.66	0.3

◊ Indicates that value overrides that in engines/tanks data base.

Cargo Mission Astrodynamics Report

Mission: BCC-EK
L02.Hum3c.ChHO.MAb.1D3cP.19s0uR.ChHO.ECCVd
L01.Car.ChHO.MAb

Reference mission: BCC-EK *Trajectory file:* A.MASE.BB.TRJ.5/9/89.Ab
Human traj: Exped. baseline traj: 2002 sprint nominal
Cargo traj: Exped. baseline traj: 2001 nominal cargo mission
Override(s) in effect: time(s) only.

Mission purpose:
TIC-1R. Final CS-2.1 Mars Exped. TEIS on MPV, time o'rides, actual ΔV's
hi boil; 2 act MAb's, 11 t 1-stg MAV, Fixed TMI & TEI tanks, 10 t down

Trajectory Type: Exped. baseline traj: 2001 nominal cargo mission

	Date	Apoapsis (km)	Periapsis (km)	Inclination (deg)	Entry Vel. (km/s)	C3 (km^2/s^2)
Earth departure (TMI)	04/15/2001	500	500	28.50		7.85
Mars arrival (MOC)	01/27/2002	300	300	76.35	6.905	23.25

Duration

	Trajectory			Override		
	Sols	Days	Months	Days	Months	
Ave. pre-TMI time in LEO						
TMIS		91.3	3.00			
MTV		91.3	3.00			
Marsbound (ETM)		286.8	9.42	430.0	14.13	Use override time
Mars orbit	0.0	0.0	0.00	0.0	0.00	Use traj. time
Earthbound (MTE)		0.0	0.00	0.0	0.00	Use traj. time
Total trip		286.8	9.42	430.0	14.13	
Car. MOC to hum. TEI		534.2	17.55	534.0	17.54	Use override time

Delta V Summary

	Delta V (km/s)		
Item	Trajectory	Override	
TMI	3.514	3.610	Use trajectory delta V
ETM DSM	0.000	0.000	Use trajectory delta V
ETM MCC	0.050		
RCS ETM	0.050		
MOC post A/C periapsis raise	0.066	0.100	Use trajectory delta V
MOO	0.100		
RCS MOO	0.050		

Mission Label Nomenclature

Launch Codes:	**Lnn**	nn = last two digits of launch year
Crew/Cargo Codes:	**Up**	Integrated mission (crew and cargo together)
	Hum	Humans on board (crew)
	nc	n = number of crew members
	Car	Cargo (unmanned)
Trajectory Codes:	**Cn**	Conjunction trajectory
	Op	Opposition trajectory
	Sp	Sprint trajectory
	LT	Low-thrust trajectory
Midcourse Events:	**--**	No major events (midcourse trim burns only)
	Dsm	Deep Space Maneuver (DSM) (major propulsive maneuver)
	Vs	Venus swingby

Examples:	L02Hum6c.SpVsDsm	Launched in 2002, Human, 6 crew, Sprint, Venus swingby and Deep space maneuvers
	L99Car.Cn	Launched in 1999, Cargo mission, Conjunction trajectory

Mission Label Nomenclature (cont'd)

Propulsion Codes:	**ChHO**	Ch = chemical; HO = hydrogen/oxygen NN = hydrazine/nitrogen tetroxide
	NEP	Nuclear Electric Propulsion
	SEP	Solar Electric Propulsion
	NTR	Nuclear Thermal Rocket
Mars Capture Codes:	**MAb**	Mars aerobrake
	MOCP	Retro-propulsive Mars orbital capture
Mars Descent Vehicles	**nD**	n = number of descent vehicles; large P denotes parachute
	nc	n = number of crew in descent vehicle
	ns	n = numbers of sols on martian surface
	nxR	n = number of rovers; x = u denotes unpressurized, x = p is pressurized
Earth Capture Codes:	**ECCV**	Earth Crew Capture Vehicle
	EAb	Earth aerobrake (for spaceship recovery)
	d	direct descent to Earth

Examples: ChHO.MAb.0D.ChHO.ECCVd Chemical LH2/LOX to Mars, with Mars aerobrake, no descent vehicles, chemical LH2/LOX for TEI, crew capture in dedicated ECCV, direct to Earth

NTR.2D.ChHO Nuclear thermal rocket for TMI, 2 MDVs, chemical LH2/LOX for TEI

V-G6.MMSS-2

APPENDIX C. HABITAT ALTERNATIVES

Several habitats were designed for Martian and Lunar Missions. Both artificial-gravity and zero-gravity designs were proposed, as well as cylindrical and disk configurations. Table C-1 lists the nomenclature adopted for describing the facilities within a habitat.

Table C-1 Habitat Nomenclature

Command and Control Center (CCC)
Work Areas
– Lab
– Maintenance
Health Maintenance Facility (HMF)
Fitness Center
Galley
Corridors
Personal Hygiene (PH)
Wardroom
Quarters
Stow
Shelter

Often a habitat will include most, but not all of the above facilities. For each habitat the total available volume inside the empty shell was calculated. From the layouts it was then possible to determine the following for each facility area:

Total floor area available: Interior volume of habitat, prior to outfitting.
Walking floor area: Open floor area.
Walkable Volume: Walking floor area multiplied by ceiling height.
Additional Free Volume: Volume above tables and beds, under desks, and of ceiling and floor storage facilities.
Outfitted Volume: Actual volume of equipment, tables, beds, exercise facilities, etc.

Table C-2 gives a summary for all of the habitats designed.

ARTIFICIAL-GRAVITY HABITATS

Both cylindrical and disk modules are considered as suitable habitats for artificial-gravity because each has distinct advantages and disadvantages. Table C-3 offers a comparison of the two types of modules.

Upon comparison, it is seen that disk modules give the maximum "floor" area for the same volume, and a compromise for the maximum longitudinal vista. The disk module also has a fall hazard, but if the acceleration levels are sub-gee, this may be more acceptable. However, disk modules do not have any potential for derivation from Space Station designs. Transverse cylindrical modules minimize the likelihood of observable coriolis effects and also packages most readily in the low L/D aerobrake configuration. In addition, an array of modules in this orientation allows a "running track" toroidal closure.

Table C-2 Martian and Lunar Habitats

	Total Vol. Available (m^3)	Total Floor Area Available (m^2)	Walking Floor Area (m^2)	Walkable Vol. (m^3)	Additional Free Vol. (m^3)	Outfitted Vol. (m^3)
MARTIAN HABITATS						
2-Cylinder; MMAG	185.0	55.5	28.6	70.1	49.0	65.9
3-Cylinder; E. Clifton	589.0	98.7	119.1	247.9	60.8	280.3
2-Cylinder; J. Danelek	545.0	181.3	149.0	364.0	101.6	79.0
2-Cylinder (short); J. Danelek	265.0	88.2	72.5	177.2	49.4	38.4
2-Cylinder; Eagle Engr.	345.0	58.8	30.5	96.5	179.2	69.3
2-Disk; J. Danelek	140.0	37.5	26.8	65.8	47.9	26.3
2-Disk; Eagle Engr.	388.0	97.4	67.8	184.7	130.7	72.6
1-Disk; E. Clifton	125.0	-	-	-	-	-
1-Disk; E. Clifton	136.0	-	-	-	-	-
1-Disk; E. Clifton	136.0	-	-	-	-	-
1-Disk; E. Clifton	136.0	-	-	-	-	-
1-Disk, (mezzanine); E. Clifton	300.0	-	-	-	-	-
LUNAR HABITATS						
1-Deck LCSV	33.5	31.2	3.6	7.6	5.6	20.3
2-Deck LCSV Habitat	88.0	30.4	9.7	24.6	13.2	50.2
Alternative LPV Habitat	85.0	31.9	10.0	25.5	12.9	46.6

Total Volume Available: Interior volume of habitat, prior to outfitting.
Total Floor Area Available: Floor area, prior to outfitting.
Walking Floor Area: Open floor area.
Walkable Volume: Walking floor area multiplied by ceiling height.
Additional Free Volume: Volume above tables and beds, under desks, and of ceiling and floor storage facilities.
Outfitted Volume: Actual volume of equipment, tables, beds, exercise facilities, etc.

Table C-3 Habitat Comparisons, Artificial-Gravity

Cylindrical, acceleration vector transverse
- -Space Station Cylinder
- -Within 1.8-meter height constraint:Area = 48 m²
  Volume = 210 m³
- -max longitudinal vista = 12.8 m
- -Coriolis effects considerations

Cylindrical, acceleration vector longitudinal
- -5 decks (Space Station Cylinder)
- -Within 2.4-meter height constraint:Area = 82 m²
  Volume = 210 m³
- -max longitudinal vista = 4.6 m
- -Corridor is ladder. minimizes corridor volume
- -Fall hazard
- -Escape easiest "down". Escape to hub requires "climbing"
- -Exercise benefit of climbing stairs.

Disk module, acceleration vector longitudinal
- -2 decks, each 25-ft x 8-ft.
- -Within 2.4-meter height constraint:Area = 90 m²
  Volume = 222 m³
- -max longitudinal vista = 7.6 m

Cylindrical Habitats

The seven artificial-gravity habitats that are discussed consist either of cylindrical or disk-shaped modules. At least two cylindrical modules are used to make a habitat. The first 2-cylinder artificial-gravity habitat is shown in Figure 3.3.1.1-1 and was designed by the Martin Marietta Astronautics Group to accommodate five people. The shape and size is based upon the space station module configuration, yet it is unique in that floors are placed perpendicular to the long axis of the module, instead of parallel. This not only allows for more total floor area than other designs, but also for greater privacy within the modules. Each crewperson has 27.7 m³ quarters.

Eagle Engineering has also designed a 5-person, 2-cylinder artificial-gravity habitat based upon the space station module configuration (Figs. C-1a and b). Both the Martin and the Eagle Engineering designs have the same total volume available (420 m³) but the floor plan of the Martin module allows for 44% more floor area, as was mentioned above. The crew quarters in the Eagle Engineering design are also 31% smaller, allowing 19.2 m³/crew quarter. However, the galley is 72% larger.

By: Eagle Engr. (L. Guerra, B. Stump)

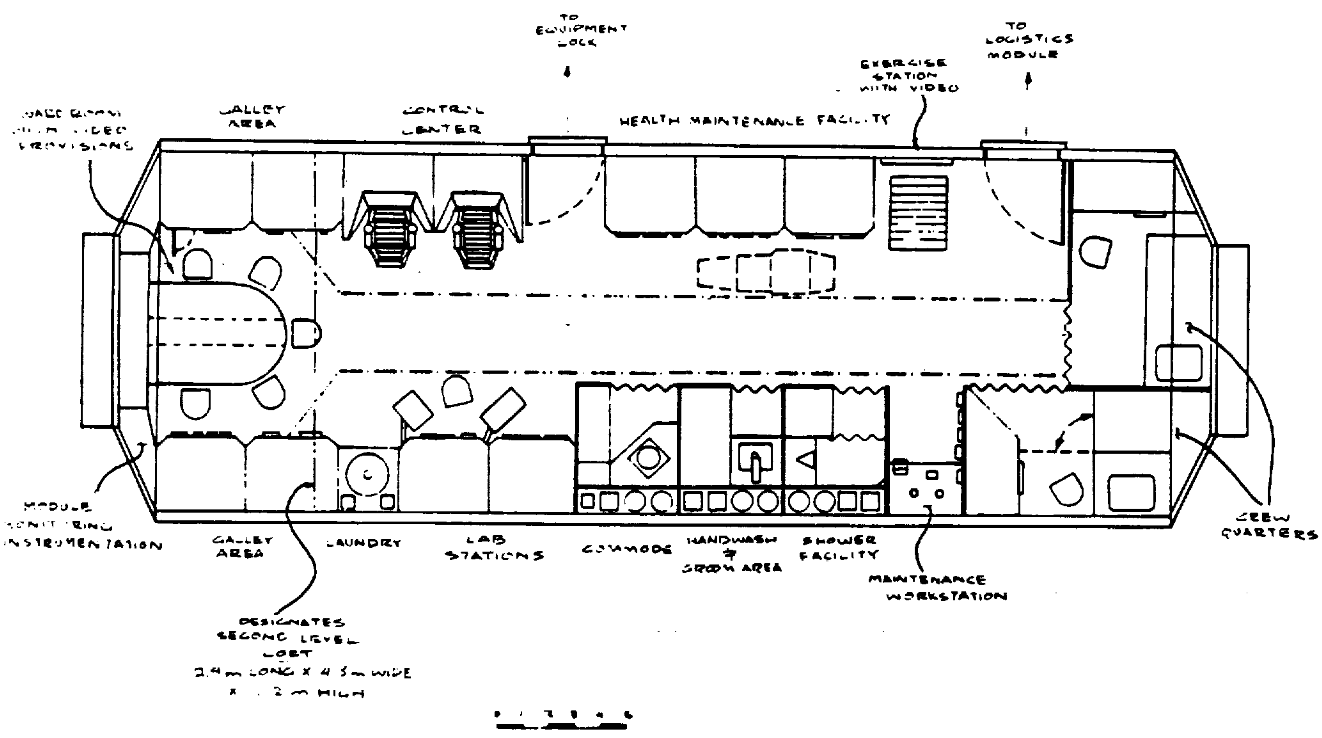

Figure C-1a Cylindrical Habitats, Artificial-Gravity

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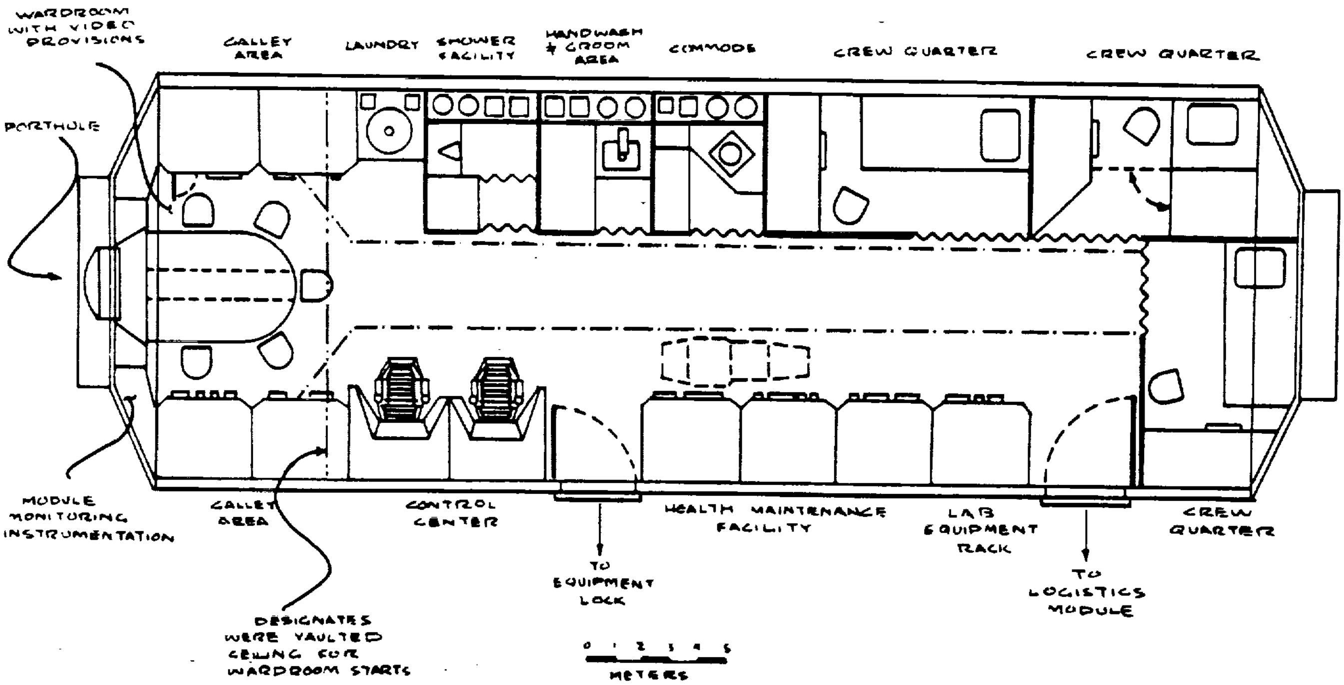

Figure C-1b Cylindrical Habitats, Artificial-Gravity

Mr. Jeff Danelek has designed the third 2-cylinder habitat, which is shorter than the other two by 5.72 m (18.75 feet). It has a crew compliment of three and is shown in Figure C-2. This design provides the most walkable volume of any of the cylindrical habitats, with 67% of the total volume allocated. This habitat has relatively 39% more floor volume than the Eagle Engineering design, yet the crew quarter's total volume is only 36.7 m³ as compared to 96 m³ for Eagle's design and 138.5 m³ for Martin's. Much of the available volume is consumed by corridors, which take up 28% of the total available volume in the habitat.

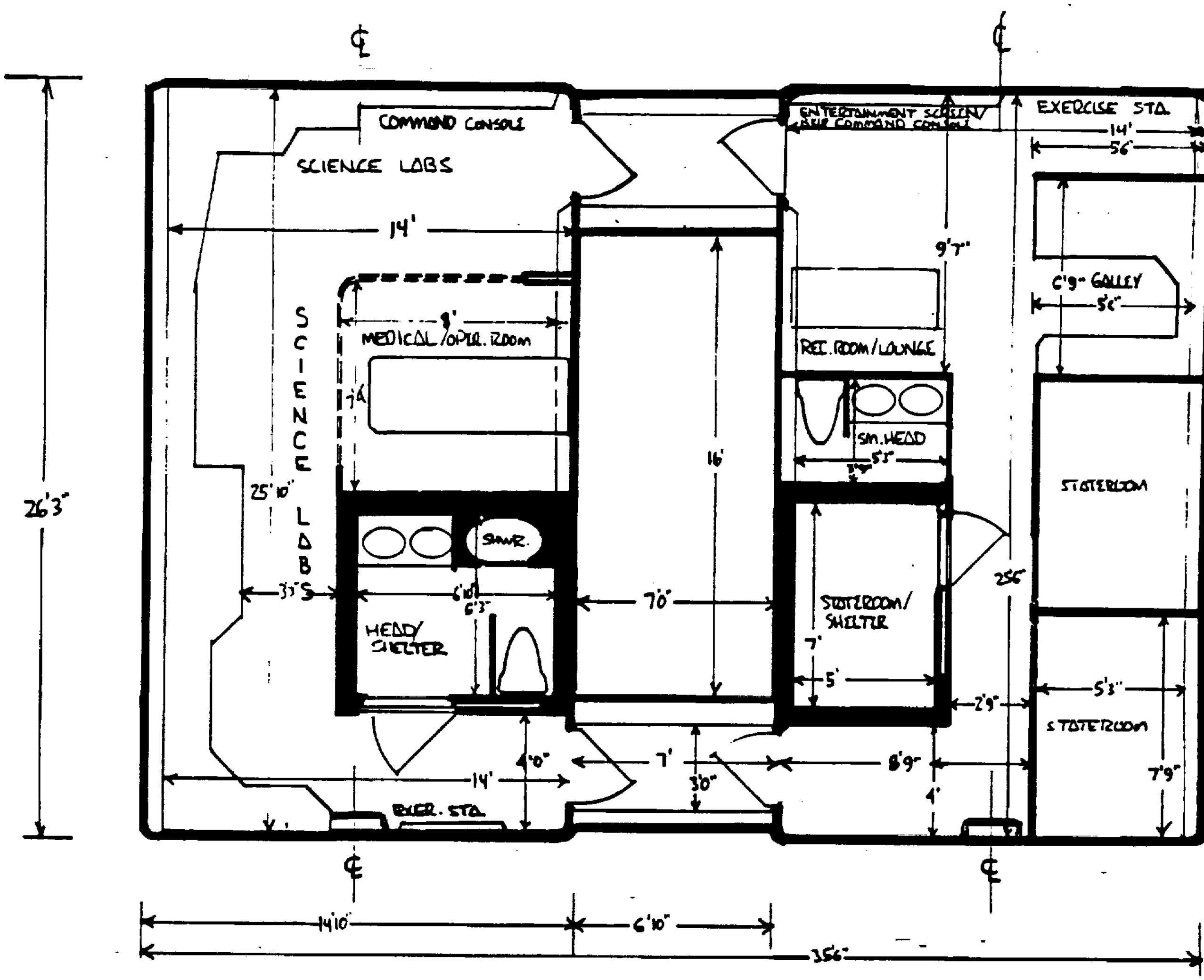

Figure C-2 Cylindrical Habitats, Artificial-Gravity

Mr. Ethan Cliffton has designed a 6-person, 3-cylinder habitat, shown in Figure C-3. Because of the third module, 210 m^3 of additional volume is now available. Of the total 630 m^3 total available volume, 42% is walkable volume, as compared to 37% for the Martin habitat, 28% for the Eagle habitat, and 67% for the Danelek habitat. The crew quarters are fairly large at 16.6 m^3 each, and storage occupies 229.0 m^3.

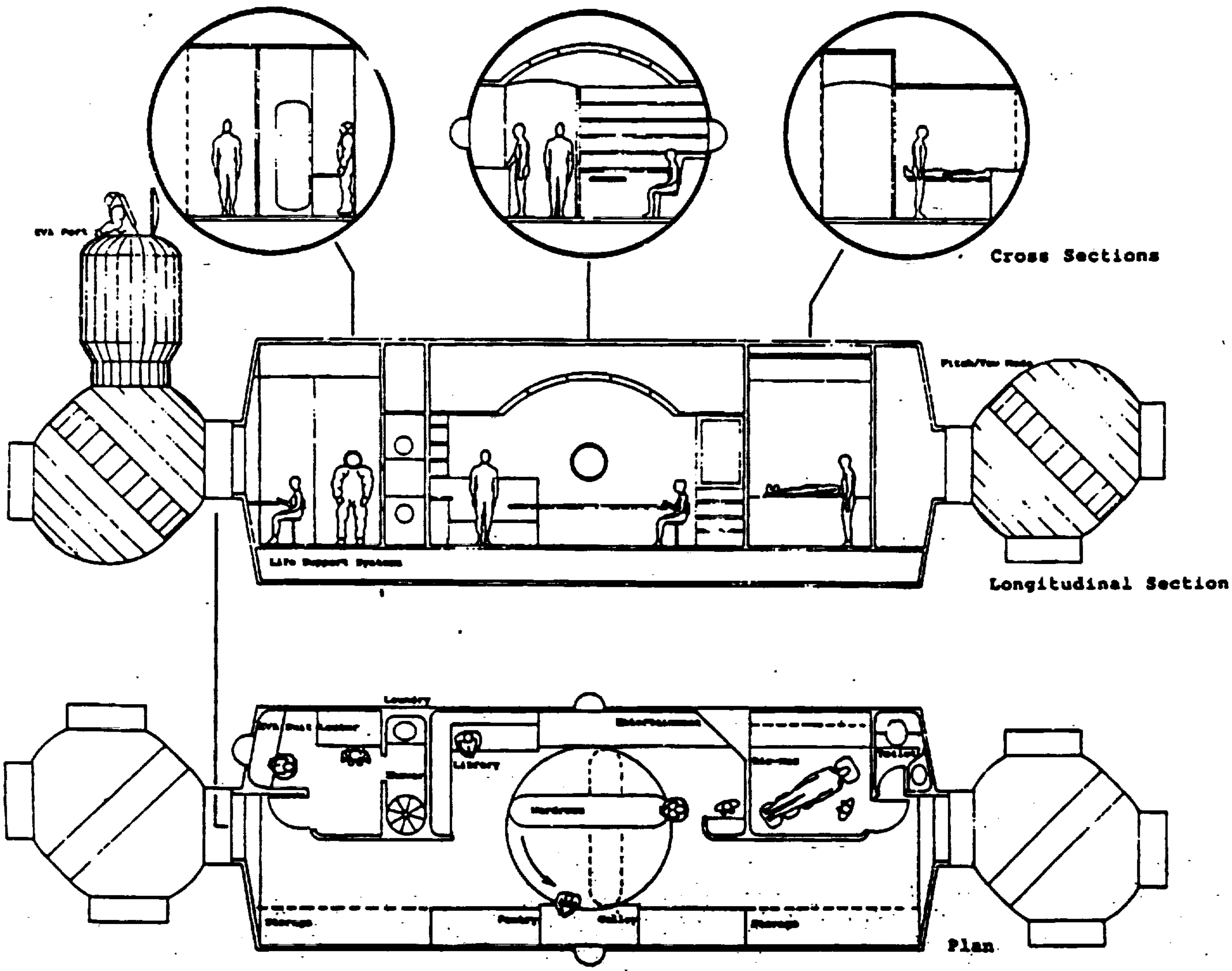

Figure C-3 Cylindrical Habitats, Artificial-Gravity

Disk Habitats

The artificial-gravity disk habitats are shown in Figures C-4-7b. The first (Fig. C-4), designed by E Cliffton, is a 136.0 m³ 4-person module. Within the habitat, each crew member has 7.05 m³ for their quarters. There is also 20.3 m³ for corridors, and 25.5 m³ for the wardroom.

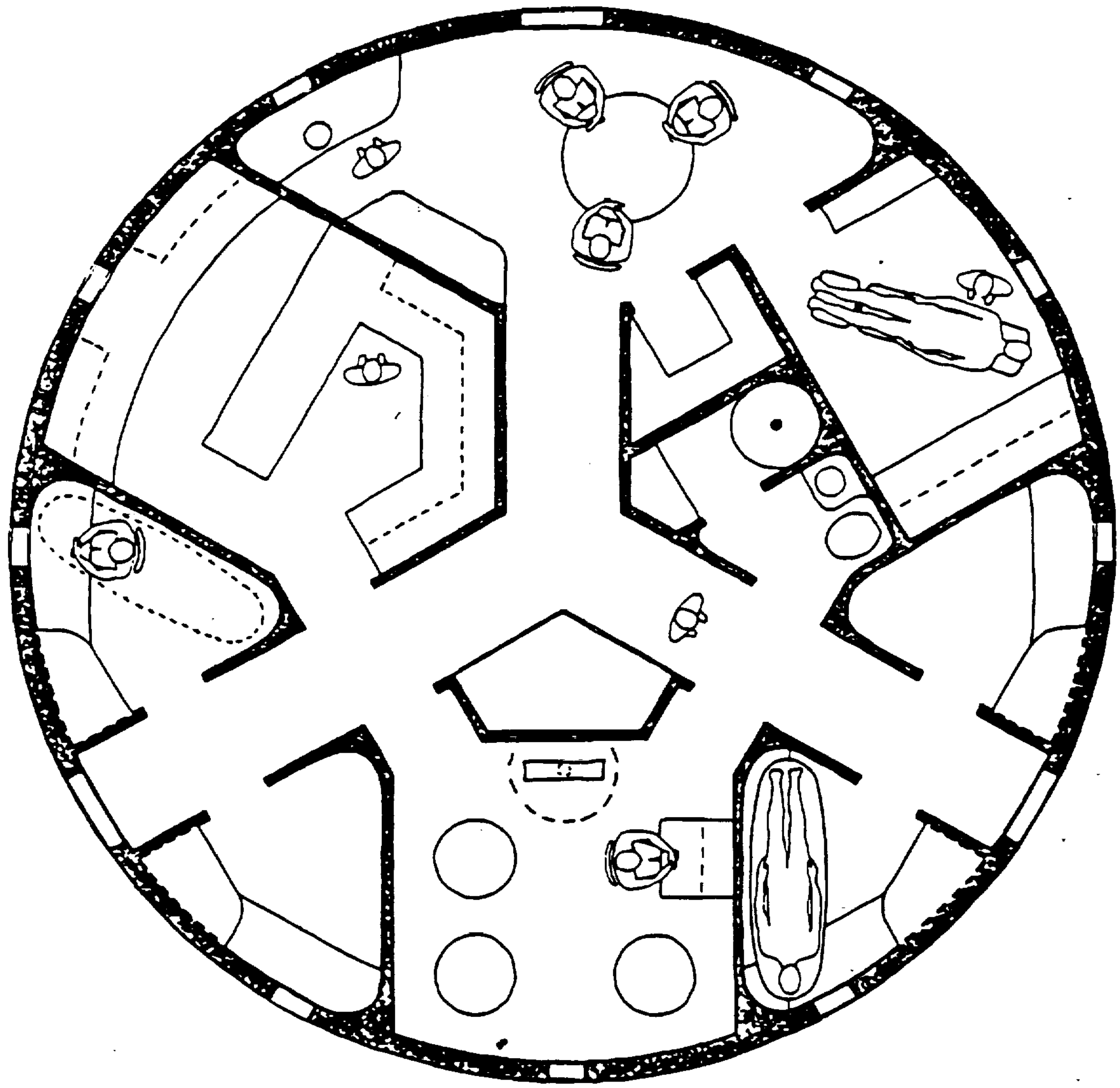

Figure C-4 Disk Habitat, Artificial-Gravity

An alternate four person design, also by E Cliffton, is shown in Figure C-5. This makes use of a mezzanine to provide for 164 m^3 of additional free volume. Here each crew member has 8.75 m^3 in their quarters. The work area is also enlarged by 74% to 80.6 m^3. Finally, corridor space occupies 35% of the total available volume, as compared to 15% for the previous habitat.

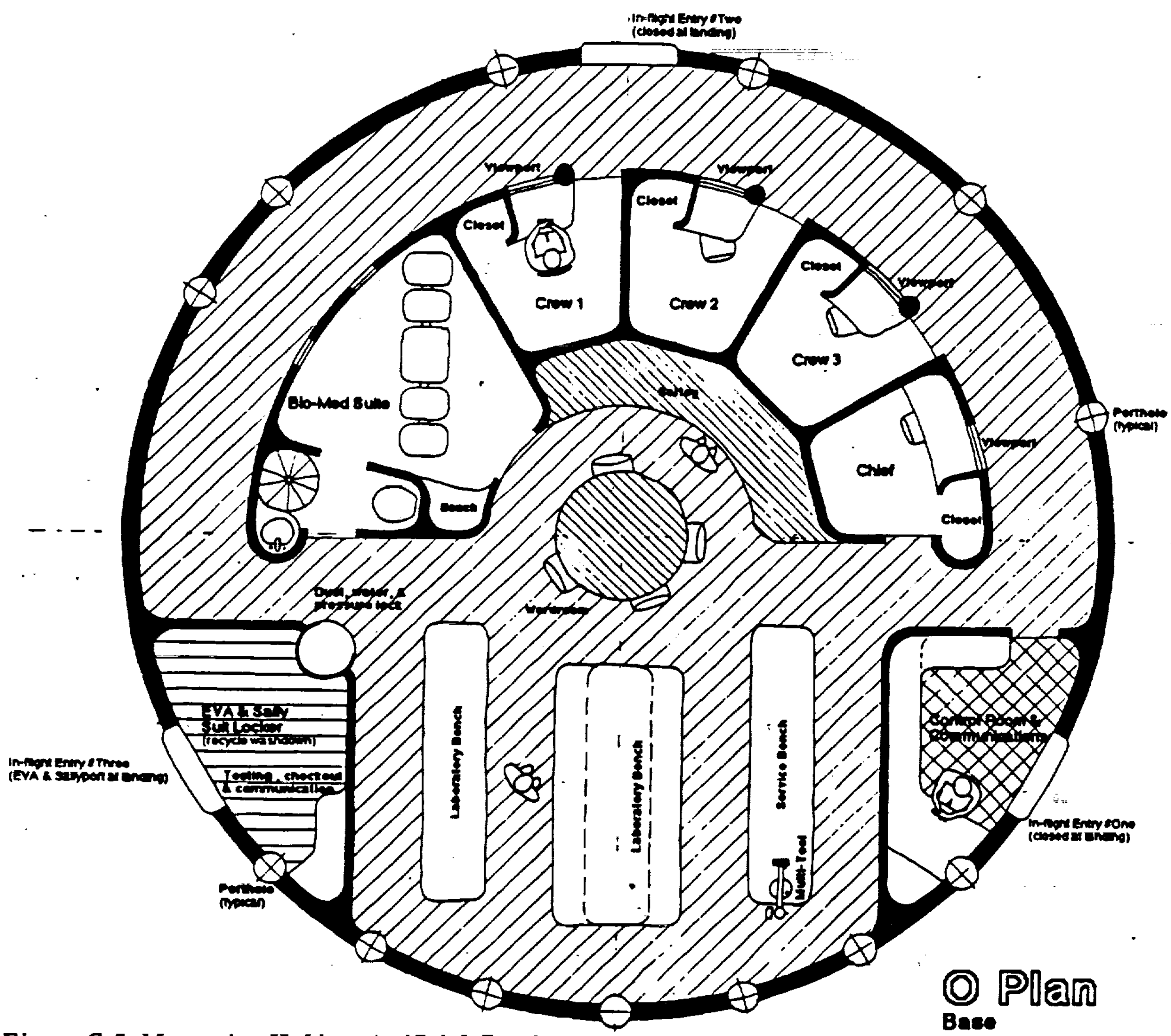

Figure C-5 Mezzanine Habitat, Artificial-Gravity

J. Danelek has designed a habitat with two stacked disks for six people (Fig. C-6). The total available volume is 225.0 m³, less than the 1-disk mezzanine design because of the low ceiling height within the disks. Each crewperson now has 12.7 m³ for their quarters, and corridor space has been cut to 6.9% of the total volume. In addition, the work area is 46.5 m³ and both the fitness area and Command and Control Center occupy 12.0 m³.

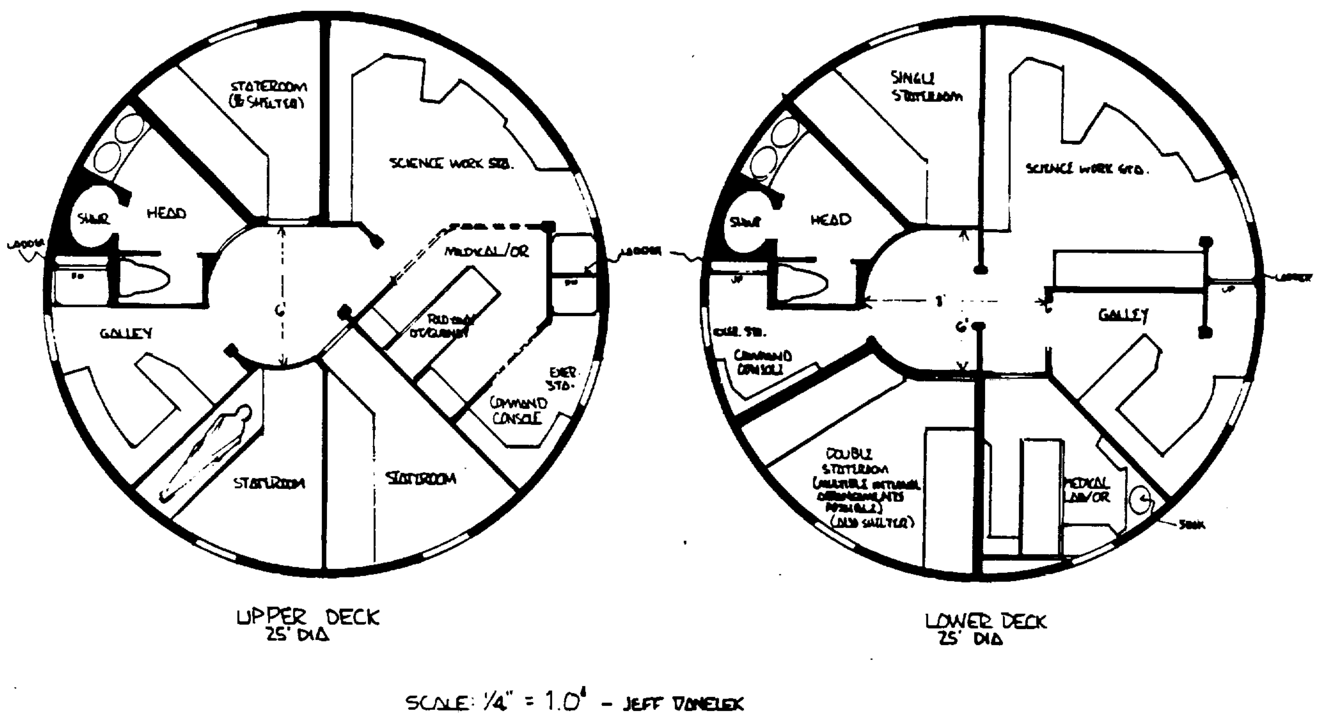

Figure C-6 Disk Habitat, Artificial-Gravity

Finally, Eagle Engineering has designed a two disk habitat which can accommodate five people (Figs. C-7a and b). Relatively spacious crew quarters (13.1 m^3) and 50.3 m^3 of storage make this the least cramped of the disk habitats. In addition, 22.9 m^3 is provided for the fitness center alone, and 55% of the total available volume is walkable volume.

By: Eagle Engr. (L. Guerra and B. Stump)

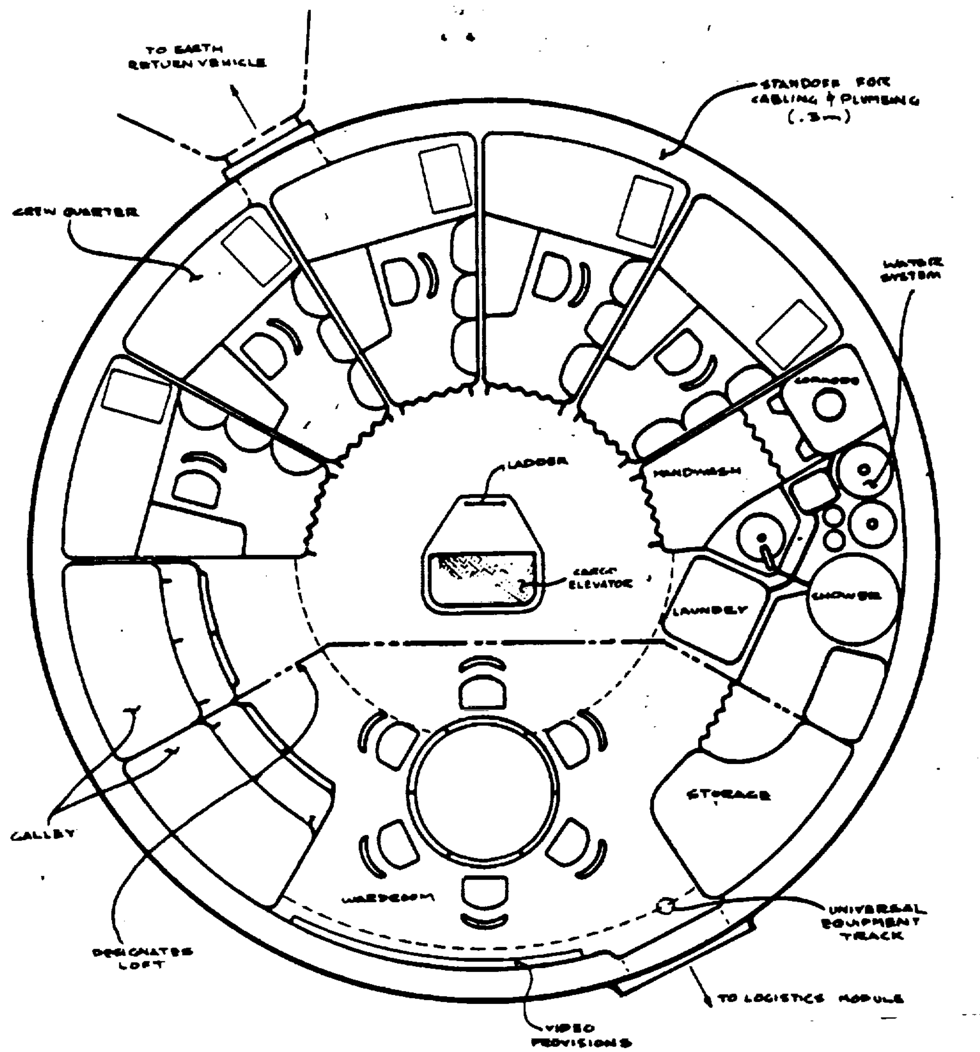

Figure C-7a Disk Habitat, Artificial-Gravity

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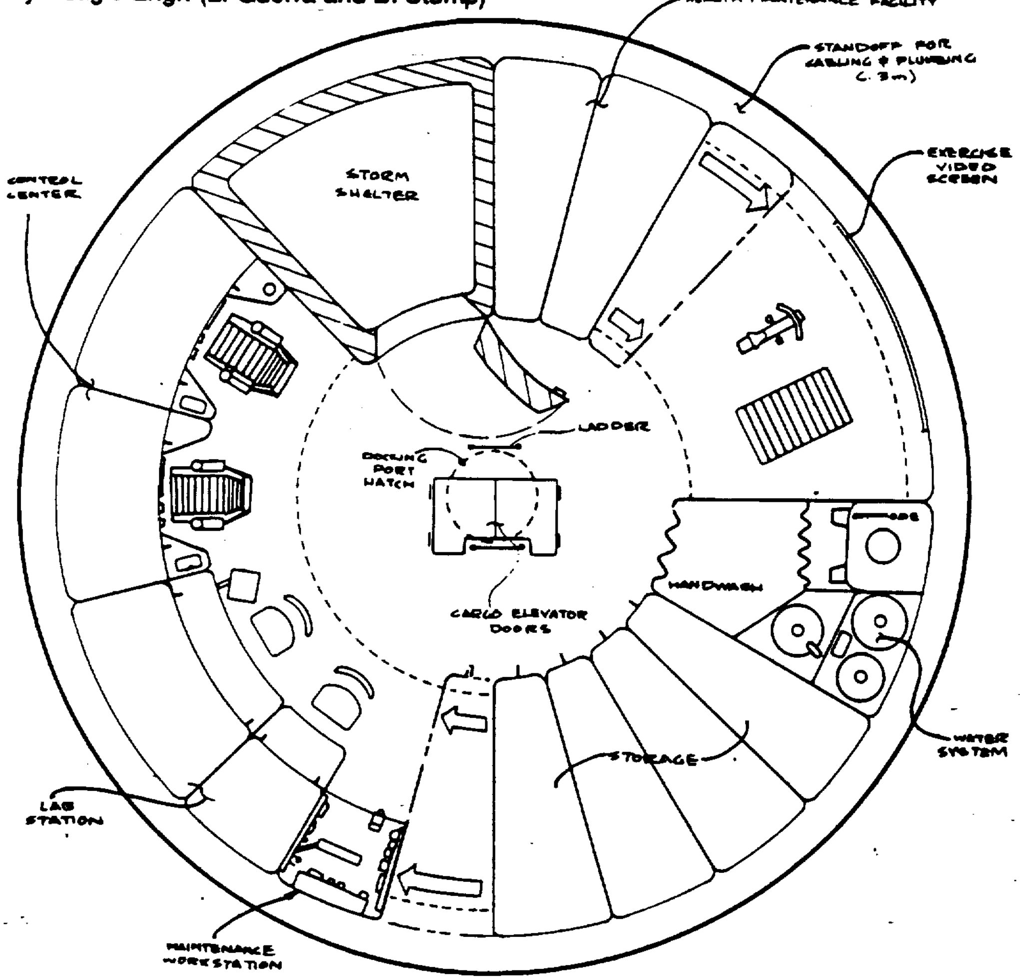


Figure C-7b Disk Habitats, Artificial-Gravity

ZERO-GRAVITY HABITATS

The first zero-gravity habitat is one designed by E Cliffton. It is the only zero-gravity habitat designed for a Mars mission. A crew of four inhabits 125 m³ on three levels, as shown in Figure C-8. The crew quarters in this cylinder are very small at 4.8 m³ each, and the work area and Command and Control Center occupy the most space at 28.6 m³. Personal hygiene has been allotted almost as much space at 24.6 m³, and the galley is the smallest of the Mars habitats at 3.6 m³

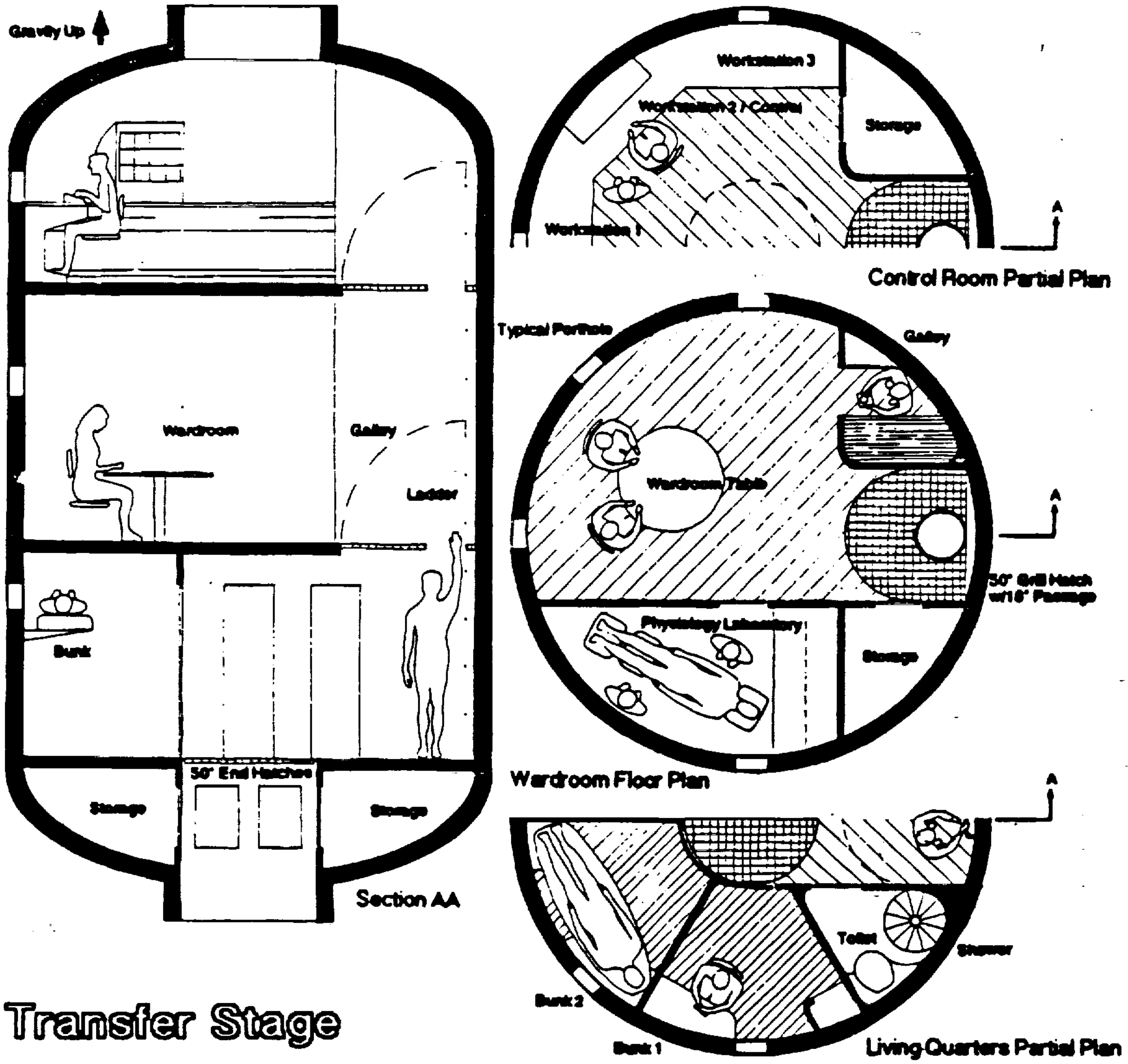


Figure C-8 Cylindrical Habitat, Zero-Gravity

The three remaining zero-gravity habitats were all designed by Eagle Engineering and are considerably smaller since they were being considered for lunar missions. The first, shown in Figure 2.3.1-1, is a 1-deck Lunar Crew Sortie Vehicle (LCSV) designed to hold 8 people. It is by far the smallest of the lunar vehicles with very little free volume. The 2-deck LCSV habitat (Fig. C-9) is also designed to hold 8 people, yet it has over twice the total available volume. It also provides a wardroom table with a galley, and has a separate EVA suit storage area. Finally, a Lunar Piloted Vehicle (LPV) is shown in Figure C-10. Like the 2-deck LCSV, it also has two floors and is designed to carry 8 people. It has approximately the same total available volume as the LCSV, and only slightly more walking space. Instead of a galley table an exercise facility is provided.

By: Eagle Engineering (L. Guerra, B. Stump)

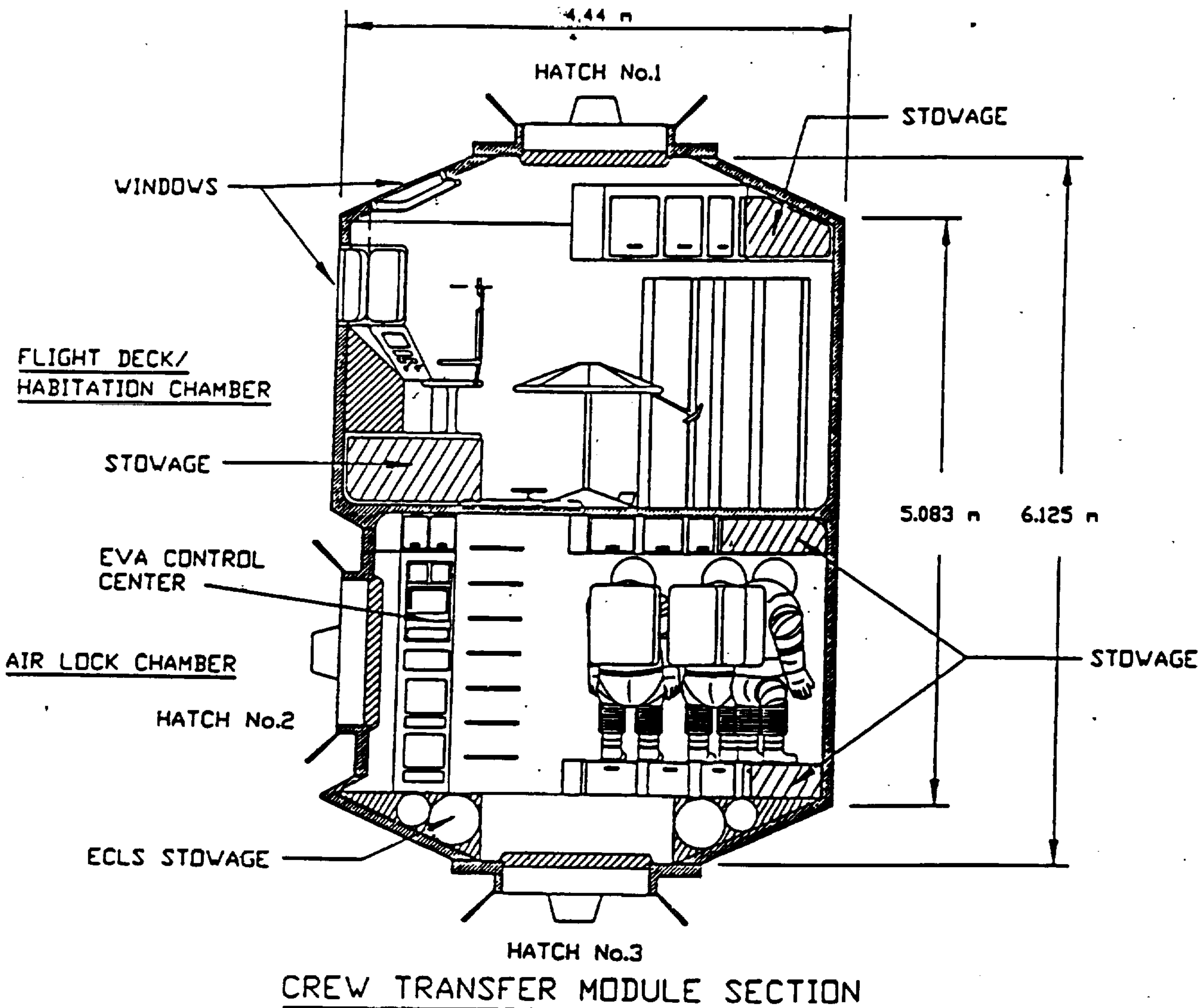


Figure C-9 2-Deck LCSV Habitat, Zero-Gravity

By: Eagle Engineering (L. Guerra, B. Stump)

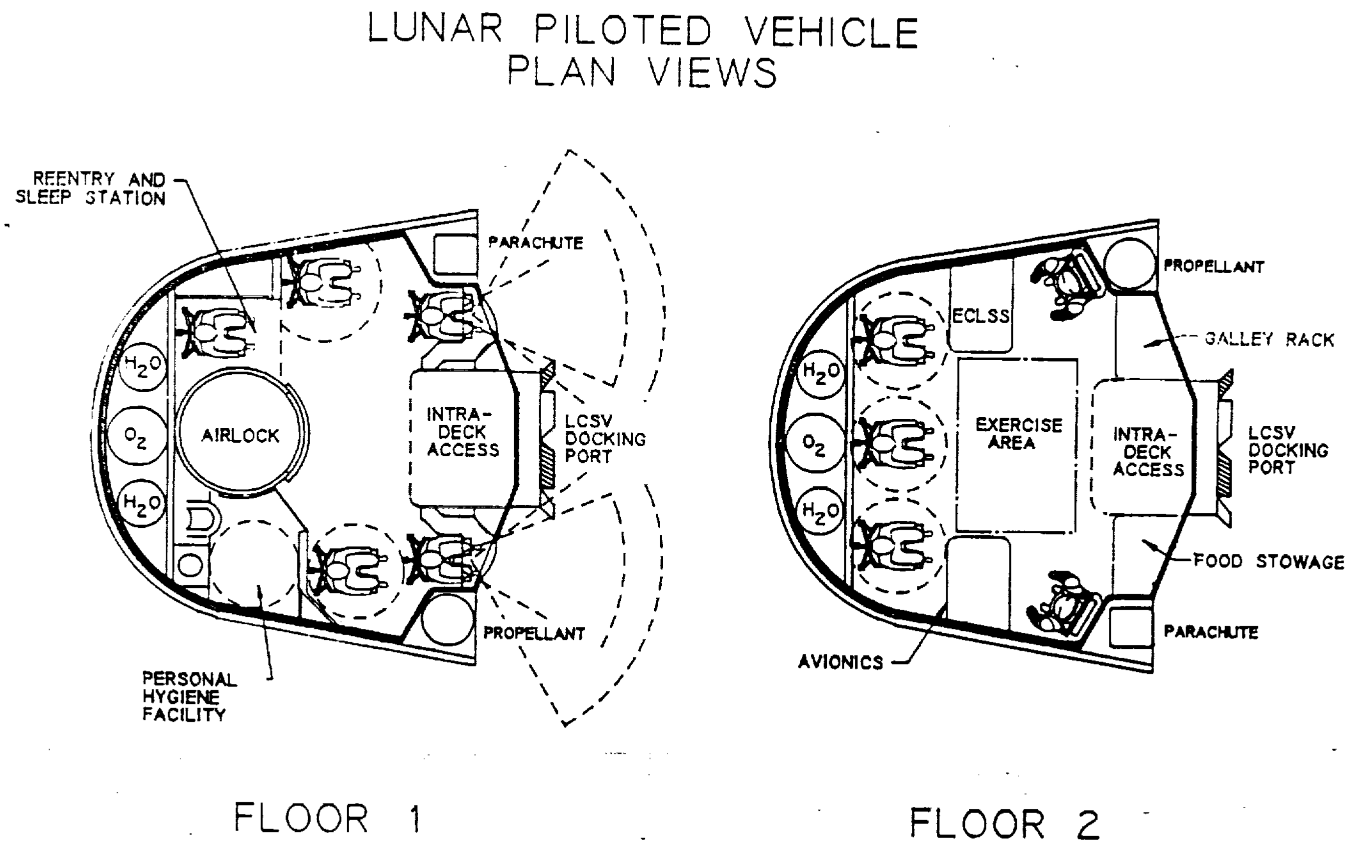


Figure C-10 Alternative LPV Habitat, Zero-Gravity

APPENDIX D. AEROASSIST ALTERNATIVES

Aeroassist is the use of aerodynamic braking in the atmosphere of a planet to reduce orbital energy. It may be applied to capture into a closed orbit from a hyperbolic encounter condition or for reduction of the size of an existing orbit. Its use in manned missions raises whole new issues in terms of man-rating requirements, but it does represent a technology that has a firm basis in the many years of entry maneuvering work performed on such programs as Gemini, Apollo, and Shuttle. At Mars, velocity reductions ranging from 2 to 6 km/s are required to capture, depending on encounter and captured park orbit conditions. For Earth capture, the delta-v's range from 1.2 to 8 km/sec with capture orbits varying from a low 1.5 hour period to highly elliptical 4 day orbit. For closed Earth orbits, the velocities vary from 2.4 km/s for GEO return to 3 km/s for lunar return. At the low end of aero energy reduction, GEO return, an aeroassist device is performance effective if its mass fraction is less than 15% of the captured payload weight. At the higher end of the scale, sprint class Mars missions can have brake weights exceeding the mass of the payload itself and still result in IMLEO far less than an all-propulsive approach (Figure D-1). Packaging considerations show preference for low L/D blunt aerobrake concepts, while higher L/D biconic shapes are attractive for g reduction in the fast encounter regimes. Manned mission aerobrakes are generally large in size (ranging from 14 m in diameter for a lunar return brake to 40 m for a Mars mission device) and thus the method of on-orbit assembly is a major concern. The primary technological areas to be considered are aerothermal, thermal protective system (TPS), guidance, navigation, & control (GN&C), on-orbit assembly techniques, and atmospheric characterization.

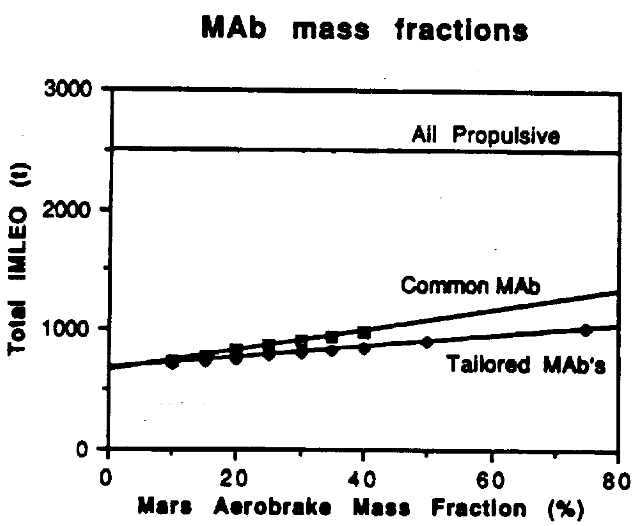


Figure D-1 Aerobrake Mass Ratio Sensitivity

The Mars Rover Sample Return represents the next major mission to the planet Mars. It is anticipated that aeroassist will play a major role in the mission with automated capture phases at both Mars and the Earth. Manned missions to Mars are hoped to be accomplished early in the next century as a major new chapter in the exploration of space by humans. The OEXP cycle 2 case studies investigated several options for manned missions. Various vehicle configurations were studied which utilized aerocapture at Mars and the Earth. Abort considerations as well as man-rating in general will be a strong driver for the design of aerocapture maneuvers for such missions.

AEROTHERMAL

Aerothermal characterization of the entry environment is crucial to correctly designing the aerobrake's TPS. Previous entry programs had extreme amounts of conservatism built into their entry heat shields because of a lack of knowledge of the thermal environment. In many cases this level of conservatism will result in marginal performance of an aerobrake. One of the biggest areas of uncertainty currently is the contribution of non-equilibrium heating. Particularly for high lift configurations flying near the skipout boundary, this can be a significant heating contribution. Currently the Aeroassist Flight Experiment (AFE) is tasked with obtaining flight data in this area for a GEO return mission. The use of computational fluid dynamics (CFD) codes should eventually reduce much of the uncertainty in characterization of the thermal environment, though there appears to be a great deal of disagreement as to how much. The impact of real-gas effects and CO_2 dissociation, while significant, does not appear to be a first order driver.

TPS

The development of advanced high temperature TPS is important for the thermal regimes as well as for aerobrake design flexibility. The use of elliptical intermediate park orbits (Figure D-2) and exo-atmospheric deceleration burns can reduce the entry energies that must be dissipated. Very high entry speeds at Earth (in excess of 13 km/s) will demand the use of ablator technology. Most high temperature ablators are inherently heavy, however, which makes the investigation of lightweight ablator technology important. Other problems inherent with ablators are their outgassing deposition onto sensitive optical/thermal surfaces and questions of multiple use because of the altered aerodynamic surface and reflectance properties. Medium temperature TPS options include derivative Shuttle tiles. Although these materials are fairly lightweight they are extremely fragile and may require new bonding techniques for extended exposure to the space environment. Multilayer metal foil or advanced carbon/carbon materials would represent more durable TPS options. In the low end of the temperature spectrum, flexible ceramic TPS such as the NASA/ARC-developed TABI (Figure D-3) can allow large diameter lightweight aerobrakes. These concepts can be automatically deployed on orbit, which reduces the assembly problem. The very significant issues of embrittlement and dynamic flutter must be investigated thoroughly, however, before these concepts can be utilized.

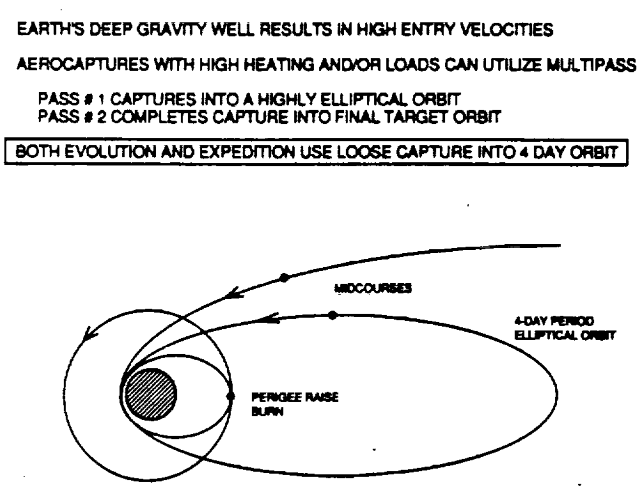


Figure D-2 Earth Aerocapture for G-Load Relief

GUIDANCE, NAVIGATION, AND CONTROL (GN&C)

The area of GN&C is critical to maintaining control of the vehicle through the atmospheric flight phase. Encounter navigation is a driver to the feasibility of the aeroassist maneuver. Uncertainty in the entry location results in rapid increases in the basic loading to the vehicle as well as rapid increases in the propagation of errors to the exit state. These errors cannot simply be "flown out" through the use of greater amounts of lift. The requirement for man-rating may maximize the use of stand-alone concepts rather than those requiring outside infrastructure. Two basic options are possible for accurate navigation state determination. The simplest is the use of radionavigation to an existing NavSat in orbit around the encountered planet. This obviously requires the development of infrastructure. Results of a Mars NavSat study are shown in Figure D-4. Very good accuracies are achieved with even late acquisition of signal. The only major technical issue involves the acquisition of navigation signals at very long ranges from the planet. This approach does levy a significant infrastructure requirement. In the case of the Earth, this infrastructure will exist in the early-1990's with the completion of the Global Positioning System (GPS) satellite network.

In the case of Mars a system of at least two satellites (for redundancy) would have to be deployed. Development of such a Mars infrastructure is more likely by the time of a manned Mars mission; its existence would be more problematical at the time of MRSR missions.

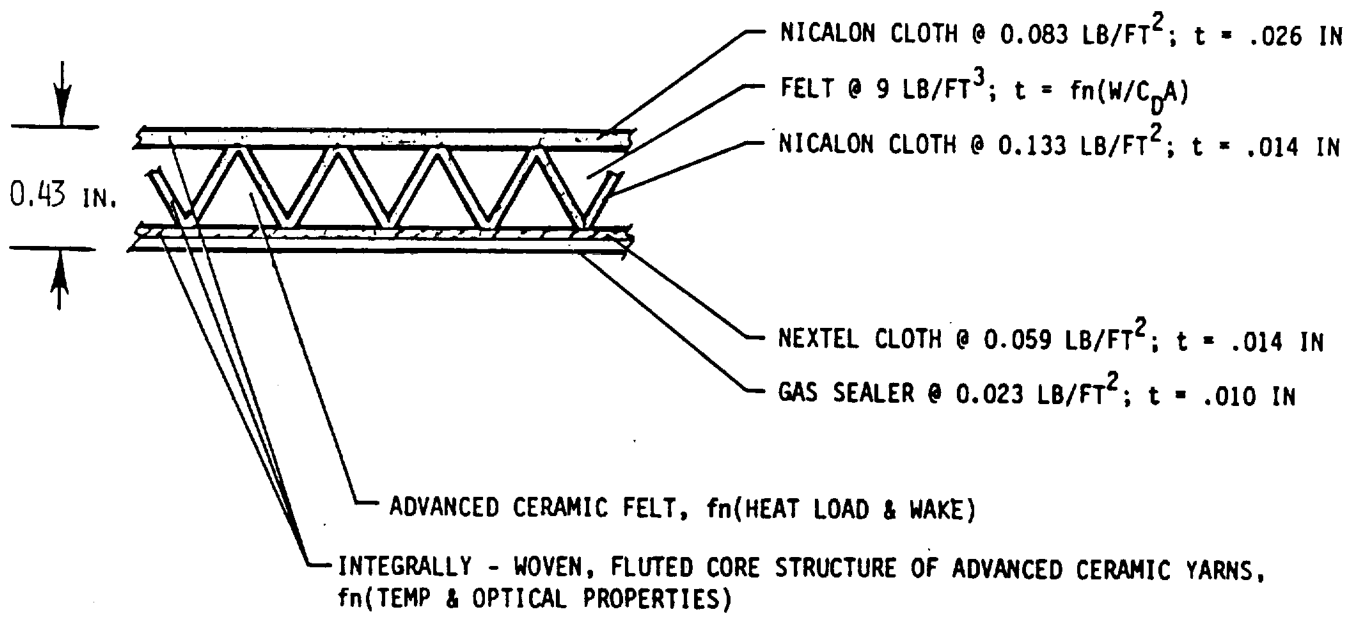


Figure D-3 Tailorable Advanced Blanket Insulation

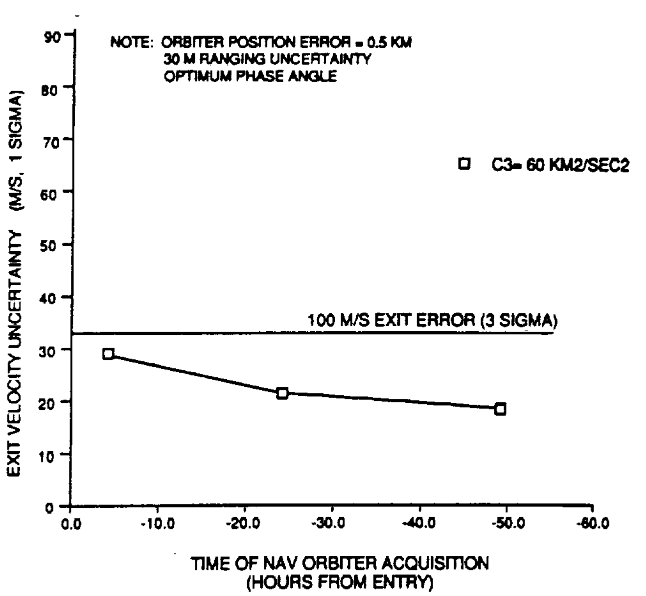


Figure D-4 Naviagation Accuracy from Cooperative Orbiter

The other navigation technique is the use of onboard optical measurements close in to the planet. This form of navigation was used by Apollo as a backup to ground-based measurements. Good accuracies are achieved with reasonable instruments if the navigation process can proceed to within a few hours of entry (this cutoff time depends on the energy of the encounter orbit). Figure D-5 shows study results using 1 arc second resolution angle measurements and a 2 km Deimos position uncertainty. Sextant type hardware with these levels of accuracy have been built and tested previously. Moon ephemeris accuracies of this order can be obtained by on-board estimation techniques or by an orbiting spacecraft's in-situ observations (perhaps even by an Earth-based Space Telescope?). Because of a lesser reliance on existing infrastructure this type of navigation might be more attractive than NavSat radionavigation for an initial unmanned mission such as MRSR. On the other hand, the autonomous requirement of long-distance unmanned vehicles means that the validation of recognition and sensing techniques for such an approach are especially demanding. Additionally, in the case of manned missions, implementation of on-board optical techniques may be required to improve man-rating by reducing dependency on external systems.

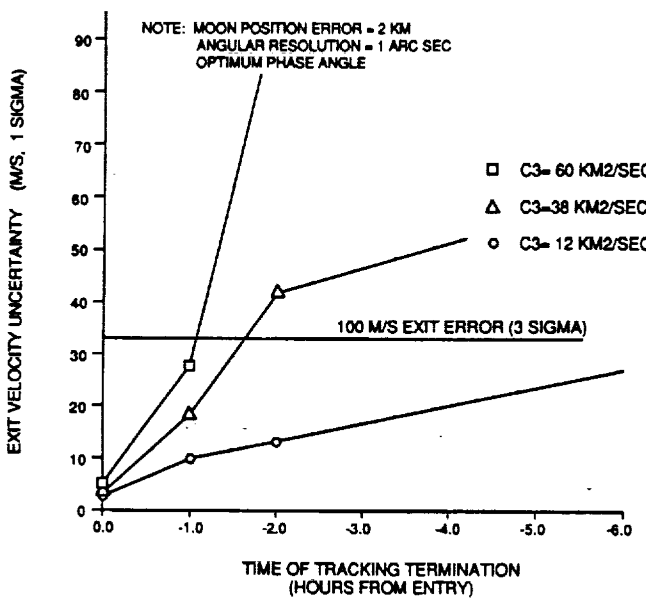


Figure D-5 Optical Navigation Measurements Using Deimos

Descent to landing presents a challenge to the navigation system in reaching a precision landing site. A variety of techniques are possible including radio ranging to an orbiting NavSat and/or ground beacon, landmark tracking, or correlation of radar terrain profiles. All the above options can provide sub-kilometer accuracy at retro-ignition to a desired landing site. The radio ranging options require the most infrastructure at the planet but are the simplest for the vehicle system to implement. Landmark tracking is the most robust stand-alone system but is also the most hardware/software intensive option for the vehicle. In addition the impact of shock refractions could make it unusable in the hypersonic phases of entry. The radar correlation option is simpler to implement and can make use of existing landing radar but does suffer from potential ambiguities in its results.

Aeroassist guidance must provide accurate end conditions while maintaining g-load and heating constraints. Low control rates are desirable both for minimum fuel consumption as well as from a crew disorientation standpoint. Adaptive guidance techniques that are responsive to changing environmental and vehicle conditions appear to be necessary. Robust algorithms that minimize extra control modes will result in a safer, simpler system overall. In the work done to date, the use of roll control of the vehicle lift vector alone is adequate to control the entry profile with acceptable exit errors. The use of atmospheric grazing passes with lift predominantly down can provide load relief for high energy missions, but TPS requirements rise due to the longer heat soak times.

The control of the vehicle in the aeropass should minimize the number of required systems in order to simplify its design. It is fairly clear at this point that the most efficient method of trajectory control is through the use of vehicle lift. Schemes for direct variation of ballistic coefficient alone are very marginal to entry dispersions. Simple rate damping and roll control of attitude by RCS jets would be a desirable goal since it provides the simplest implementation. The onboard guidance system should be able to handle a certain amount of uncertainty in the vehicle angle of attack but this cannot be excessive for heating as well as control reasons. The vehicle angle of attack is impacted by uncertainties in the aerodynamic properties as well by shifts in its center-of-gravity (cg) location. The aerodynamic properties can be determined by a combination of wind tunnel and CFD testing, probably to within a resulting ±1° angle of attack. If it is assumed that the net angle of attack is desired to be within ±2°, then the cg tracking's contribution must be maintained within ±1°. Figure D-6 shows vehicle envelopes for a blunt cone's cg with this constraint. Although the cone is relatively insensitive to longitudinal cg shifts, the lateral location requirements are quite tight (11 cm for a 39 m diameter aerobrake). Analogous, but inverse oriented results are obtained if a biconic entry vehicle is used, Figure D-7.

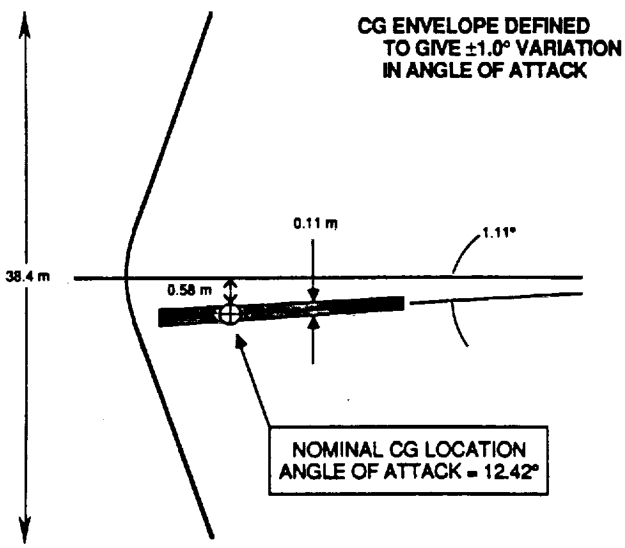


Figure D-6 Low L/D Aerobrake–cg Requirements

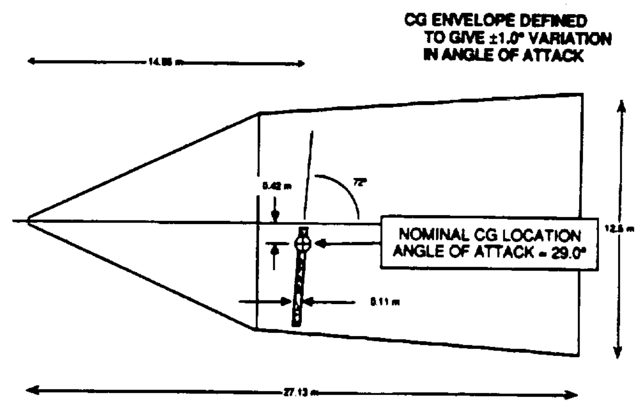


Figure D-7 High L/D Aerobrake–cg Requirements

Because manned vehicles must fly in deep space for extended periods of time with crew motion, consumables relocation and main propellant boiloff, the centering of the vehicle cg at the time of entry will be a significant issue. Relocation of personnel and habitat items to predetermined locations prior to entry will be required as a first step. Stowage of antennae and any other low-g equipment will have to be accounted for to first order in the overall cg design of the vehicle. The control of propellant location will be a significant a design driver requiring the use of propellant baffles and/or traps perhaps combined with settling burns prior to entry. Propellant cg control within tankage is a significant issue for any propulsive vehicle but design solutions do exist. The use of calibration roll maneuvers can be undertaken which measure the cg location via the use of strategically placed accelerometers. The sensitivity of these instruments is not unreasonable for vehicle roll rates in the 10-20 deg/s range. Pre-entry correction of the measured cg location can then be accomplished by shifting movable mass. Finally, the vehicle attitude can be actively controlled in the aeroassist phase, either through the use of flaps or by the actuation of a control mass (inert or propellant). This last option is clearly the most mechanism intensive but may be required for some configurations.

AEROBRAKE ASSEMBLY ON-ORBIT

For near-term unmanned missions such as MRSR the current emphasis is to utilize aerobrake and packaging concepts that do not require any on-orbit deployment or assembly. This is made possible by the large payload diameter capability of the Titan IV and Space Shuttle launch vehicles as well as the relatively small size of the mission spacecraft. However, because of the large size of manned missions, their aerobrakes may require significant on-orbit assembly and preparation for flight. Various concepts have been developed in the course of OEXP and previous studies. The use of advanced launch vehicles with large (12.5 m) payload diameter capability would enable the use of large biconic aeroshells to be delivered to orbit intact. This solution was used in the OEXP Mars Expedition case study where the payload to be delivered to Mars orbit was reduced in size by the use of a separate unmanned cargo flight. Although this monolithic approach obviates the need for on-orbit assembly it does present a driver for launch vehicle evolution. Because of their obvious packaging flexibility, several concepts of blunt cones were investigated as well. The use of flexible ceramic TPS concepts enable deployable aerobrake concepts (so-called flex/fabric aerobrakes) which thus eliminates brake assembly (but not the subsequent outfitting of mission elements onto the aerobrake assembly). These aerobrakes could be remotely deployed with springs and latches after delivery to orbit similarly to furlable space antenna designs (Figure D-7). The critical technologies to enable these forms of space-deployable aerobrakes involves the development of high temperature flexible TPS as mentioned above. These concepts were utilized in the OEXP Mars Evolution and Lunar Evolution case studies.

Finally, a rigid space-assembled blunt aerobrake concept was developed as an alternative to the flex/fabric aerobrake used in the Mars Evolution case study. This concept made use of advanced carbon/carbon high temperature material with backing insulation to prevent re-radiation. Advanced carbon/carbon has very high thermal flux capabilities (around 100 BTU/ft2/s) as well as good structural capability so that it eliminates the need for separate structural & thermal systems on the front of the brake. This avoids tile bonding problems as well as allowing a minimum diameter because of its high thermal capability. The minimum thermal diameter for the Mars Evolution vehicle is about 70 ft. However, because this presents severe packaging problems a diameter of 100 ft. was used as a design point. This concept was then utilized as a reference case for on-orbit assembly.

In the case of space assembled aerobrakes the number of pieces as well as the complexity of their integration should be minimized. These goals are usually in conflict with each other. Above all, the concepts for assembly must be self-reinforcing under air load if at all possible to provide fail-safe features as well as reducing the complexity of the joint interface. Such a concept is shown in Figure D-8. Using the concept of a central core section surrounded by assembled petals its conical form tends to give compressive forces at all seam lines. The use of shear blocks mating into slotted guide receptacles can simplify the alignment process (Figure D-9). Flexible high temperature seals can be located in the recesses of grooves which will alleviate their thermal flux requirements. Because the joints are under compression with shear blocks taking up the majority of the interface load, the fasteners that hold the panels together can be minimized both in size and number. Depending on the level of infrastructure available at the on-orbit assembly facility these fasteners can either be engaged by an external device (e.g., a drill on the end of a remote manipulator) or by a device that remains with the brake (actuator motors permanently mounted into the petals themselves). The first approach would minimize aerobrake scar weight at the expense of facility complexity while the second would do the opposite.

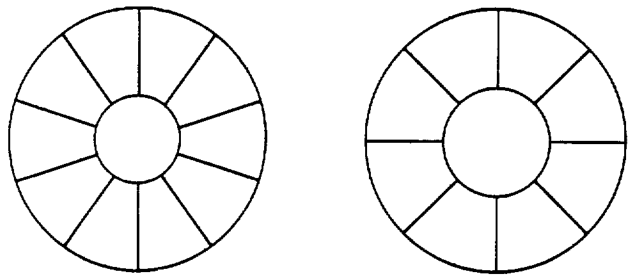


Figure D-8 Rigid Low L/D Aerobrake for On-orbit Assembly

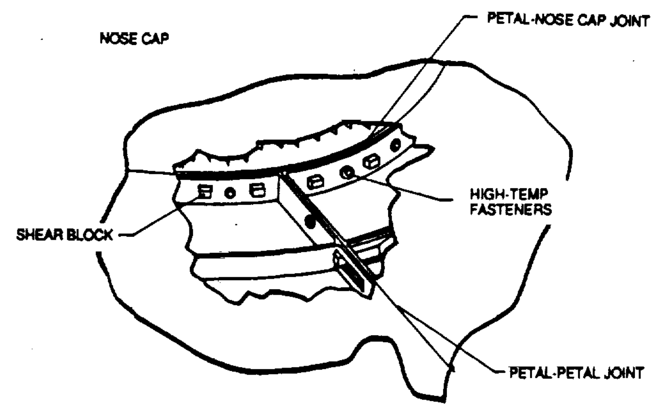


Figure D-9 Aerobrake Joining Concepts

The complexities of assembly of large structures on-orbit are significant and require extensive operations analysis. The problems associated with zero-g kinematics of large units apply both to human space walkers or teleoperated robots. The use of robotic and teleoperated assembly units must be maximized because of the great cost of EVA. Differential expansion due to severe solar heating can probably be minimized through the use of low expansion coefficient materials such as composites. Self checkout of structural integrity will result in significant instrumentation such as contact switches and strain gauges. Inspection of the finished structure will involve such techniques as differential laser interferometry to verify smoothness of fit. Finally, a low entry-velocity flight test into the Earth's atmosphere may be required as a final overall system check. The assembly operations for the Space Station Freedom will certainly drive solutions to many of these issues. A good technology base currently exists for high temperature joints and seals for the shuttle with its payload bay, landing gear and External Tank umbilical doors. However, this needs to be extended to higher heating rates.

ATMOSPHERIC UNCERTAINTIES

One of the first-order drivers to the design of the aerobrakes will be the degree to which the Mars atmosphere's uncertainties can be reduced. The bulk density variation of the atmosphere is a first order driver to the required L/D as well as the thermal margins because it alters the density altitude at which the vehicle must fly. Density shears and gravity waves present fluctuating atmospheric conditions which drive the control rates and exit errors of the vehicle. Table D-1 shows the impact of an unpredicted density dispersion upon the exit apoapsis accuracy for a representative manned Mars aerocapture. The small scale structure of these density phenomena (which affects dynamic loads) cannot be predicted but the larger scale structure (which impacts exit errors) should be, depending on the investment in in-situ observations. Dust storms alter the density structure of the atmosphere by solar heating of the optically thicker gas. However, much of this shift can probably be accommodated with far-encounter observations. The impact of winds will be felt most strongly in the entry to landing phase and its need for precision landing. Better characterization of the long-term behavior of the Martian atmosphere is called for with a dedicated orbiter that can make observations down to 20 to 40 km in altitude with good resolution (5 to 10 km).

Table D-1 Impact of Density Dispersion Exit Contditions

DISPERSION	EXIT ORBIT PERIAPSIS (KM)	EXIT ORBIT APOAPSIS (KM)	ΔV TO REACH PARK ORBIT (MPS)**	PEAK LOADS (g's)
NOMINAL	37.6	33845.4	12.3	6.91
LOW PRES ATMOS	41.0	36129.8	23.4	6.89
HIGH PRES ATMOS	34.9	31323.2	28.1	6.39
VIKING 1 ATMOS	45.4	35533.1	20.3	6.78
VIKING 2 ATMOS	32.7	31460.8	27.3	6.54
ΔPER = +2.78 km*	37.7	33844.7	12.3	6.65
ΔPER = -2.78 km*	37.8	33854.2	12.2	6.76
Δ ALPHA = +2.0°	37.9	33853.7	12.2	7.01
Δ ALPHA = -2.0°	37.4	33846.9	12.3	6.82
RMS OF DELTAS FROM NOMINAL	3.6	1585.7	9.1	0.26

* DELTA FLIGHT PATH ANGLE = ±0.10° (AT 125 KM)

** FINAL PARK ORBIT OF 250 X 33851 KM IS REACHED VIA ΔV1 AT APOAPSIS FOLLOWED BY ΔV2 AT PERIAPSIS (ΔV=ΔV1+ΔV2)

PROGRAMMATICS

Currently the AFE program is planned to better characterize aeroassist issues associated with non-equilibrium radiation, surface catalysis, and flowfield characterization via a flight experiment in the 1994 time frame. The flight data obtained will also act to validate CFD codes. This experiment will investigate the speed regime consistent with return from geosynchronous orbit (entry speeds of 9.6 km/s) of an Aeroassisted Space Transfer Vehicle (ASTV). The AFE flight test database will probably be sufficient to enable the development of an unmanned Lunar Evolution cargo vehicle. This vehicle could then be used to validate entry configurations for the subsequent manned vehicles. On the other hand, if the manned and unmanned flights are concurrent or if the advanced technology aerobrake concepts (such as flex/fabric) are used, a flight of will be required.

For manned planetary flights, one to two further flight tests will be required depending on the type of aerobrake configuration utilized. In all cases it would be highly optimistic to assume manned aerocapture at Mars could be undertaken without first accomplishing a precursor mission. Such a mission as the MRSR could fit this requirement though it would need AFE levels of instrumentation for suitable data return. Because such a mission would need to be completed before the start of serious manned Mars designs it would have to be completed (including data reduction) at least five to six years prior to launch. The current OEXP case studies looked at initial flights in the 2004 time frame, which would present a severe schedule driver. If high speed entry at Earth is maintained as a mission requirement (primarily driven by sprint mission and powered Mars abort options) it will probably necessitate a dedicated Earth capture flight test due to the very high radiation and leeside heating regimes encountered. Although the MRSR return segment could theoretically be boosted in energy to accomplish this mission, it is more likely that it would be carried out as a dedicated Earth orbital experiment in order to avoid jeopardizing the return of the MRSR's samples. Finally, if the use of the flex/fabric aerobrake concept is selected it will require a flight verification of the design issues mentioned earlier, probably replacing the high speed Earth capture test since the two requirements are mutually exclusive. Here again, for the baselined mission start dates used in OEXP case studies, the high speed Earth capture test and/or flex/fabric aerobrake flight test would have to be completed by 1999. This date is by no means technically impossible but it does require very aggressive funding to be achieved starting in the very early 1990's.

APPENDIX E. DETAILED REQUIREMENTS
SRD REQUIREMENTS AND SPECIFICATIONS ANALYSIS

Definitions:

Groundrules.	Broad rules set out in the SRD, either for overall application or to the particular Case Study
Requirements.	Detailed technical specifications given in the SRD to specify or constrain the Case Study.
Deviations.	Exceptions that must be taken to the SRD as written.
Derived Requirements.	Additional requirements that can be deduced from one or more (e.g., via combination of) SRD Requirements.
Assumptions.	Technical specifications not given in the SRD. Assumptions are generated by the TIA in order to conduct the Case Study.
Partials/Sensitivities.	Parametric variations about the selected point design, specified by SRD *or* TIA.
Options.	Changes from the baseline set of SRD requirements which uniquely define a Case Study, but are clearly specified in or inferred from the SRD as additional choices to be studied.
Alternatives.	Several different approaches to the same general objectives may be considered in implementing the Case Study. Each Alternative is generated by the TIA by replacement of one or several of the SRD requirements.

LUNAR EVOLUTION

Extracted from signed SRD dated March 3, 1989, Section 2.2 Lunar Evolution Case Study (pp. 7-47)

Groundrules:

- Achieve a test bed and learning center for long duration planetary missions (2.2.1.A)
- Develop a significant science research capability (planetary, astronomy, life sciences) (2.2.1.B, 2.2.2.G, H, 2.2.3.1.B)
- Develop resource potential of the moon (2.2.1.C, 2.2.2.F). Propellant production (2.2.3.1.H)
- Develop a lunar gateway for Moon and solar system (2.2.1.D). Mars evolution (2.2.2.K)
- Control ETO delivery per year to a specified level and determine capability at Moon (2.2.2.B)
- Reusable vehicles (2.2.2.I, 2.2.3.1.D)
  Exception: Maintain an expendable, post-TEI contingency direct Earth entry capability (2.2.4.2.2)
  Trans-lunar vehicles are assembled and serviced at Station Freedom (2.2.3.1.E)

Lunar ascent/descent vehicles are serviced and maintained on lunar surface (2.2.4.2.1.3)
- Annual IMLEO limit of 570 t/yr to 500 km gross, ≤ 90 t/yr dry (averaged over 2 yrs) (2.2.4.1.1)
- 3 phases: Outpost/Human-tended (crew 4, 6-mo. TOD)
  Experimental (crew 8, 2-yr TOD)
  Operational (crew of 8 up to 30, 2-yr TOD) (2.2.4.1.3)
- Commonality: minimum number of vehicles to fulfill functions of operational phase (2.2.4.2.1.2)
- No orbital nodes other than Space Station Freedom (2.2.4.4)
  Minimize requirements for on-orbit assembly, but
  make appropriate use of Freedom Station in time period 2004-8 (2.2.4.2.3.4.3).
  Freedom will *not* be used to store main-stage propellants [e.g., H/O] (2.2.4.3.2.8)
- Technology Level 6 for EAb by 1996, NEP by '06, LLOX production by '98, LH2, metals (2.2.4.1.4.1.L) by '08,
  1 MWe surface power by '98, lunar construction and transportation by '01 (2.2.4.9)

Requirements:

- ETO Cargo (2.2.4.8): Launches ≥45 day intervals, 4 launches/yr.
  140 t to i=28.5°, 500 km LEO. Accommodates 12.5 m diameter x 25 m long cargo load.
  Supports first mission in 2004.
  ETO Crew (2.4.4.8): 2 launches/yr. Not more than 3 days after a cargo launch.
  4 crew, 5 t cargo for lunar mission. 6 person servicing/repair crew plus 5 t cargo.
  Provide ferry in vicinity of Freedom [is this an OMV?] (2.2.4.8.1.2.3)
  Provide rescue capability (lunar STV) for 8 crew (2.2.4.8.2.1.4, -2.2.4)
- Program initialized in 2004 (2.2.2.D, 2.2.3.1.A, 2.2.4.1.2)
- Crew 4, growing to 30 (2.2.3.1.C) at base
  Minimum crew 2; Two are IVA/EVA proficient, two are EMT (2.2.4.1.6.1)
- Transportation vehicles shall be sized for pre-set ΔV's and durations (2.4.2.2.2.1, -3.1, and -4.1)
  Crew vehicles accommodate 8, mixed gender (2.2.4.2.2.2, -4.2), with 2 t cargo (-.3, -.3)
  LEO<—> LLO, Cargo of 20 t for initialization, TBD t for earth to LLO, 2014 (2.2.4.2.3.3)
  LLO<—> LSurf, Cargo of 20 t for all phases (2.2.4.2.5.3)
  LLOX, LSurf—>LLO. Amount of LLOX is 0.5 that needed for roundtrip
  of cargo or crew, LLO—>LEO—>LLO, initialization . TBD t for 2014 (2.2.4.2.6.3)
- Chemical propulsion, at initialization (2.2.3.1.G). H/O for crew (2.2.4.2.2.4.1, -4.4.1)
  H/O for cargo at initialization; <u>NEP</u> at technology enhancement (2014) (2.2.4.2.2.3, -3.4)
  H/O for LLOX transportation (2.2.4.2.6.4)
- Powered flyby aborts at moon (2.2.4.1.5)
- 30.3 d roundtrip flight time to moon (initial capability) (2.2.4.2.2.1.A, -3.1.A)
- Aerobraking at Earth for return vehicles (2.2.3.1.F, 2.2.4.2.2.4.2, -3.4.2).
  Entry speeds ≤ 11.5 km/s (2.2.4.1.5)
- Crew work/rest scheduling
  6 duty days/wk @ 8 hrs duty time/work day; 2 hrs/day exercise (2.2.4.1.6.2)
- Dual failure tolerant subsystems, operable from redundant locations (2.2.4.1.6.4.3)
  [<u>Note</u>: TIA assumes one engine out for cryopropellant-based propulsion and excludes this requirement for storable bipropellant-based systems]
- Protect against excessive cosmic and solar radiation.

Capability accessible within 30 minutes (2.2.4.1.6.4.2)
[Note: Only protection against solar flare radiation will be provided by TIA]
Radiation protection on MPV: provide 5 g/cm2 shielding (2.2.4.2.2.4.4)
[Assumption: does not include slant-path or astronaut mutual shielding benefits]

- All hazardous materials stored outside of pressurized elements (2.4.4.1.6.4.4)
- Isolation and rapid egress from any habitable element in emergency (2.4.4.1.6.4.5)
- All habitable elements shall have redundant escape paths (2.4.4.1.6.4.6)
  [but not necessarily rapid. See use of EVA escape path, under Derived requirements]
- Pressure integrity checks; adequate day/night lighting; pressure hatches (2.2.4.1.6.4.7-9)
- EVA-suited operation of emergency controls (2.2.4.1.6.4.10)
- 60 d safe-haven capability *on the lunar surface* (2.2.4.1.6.4.11)
- Para. 2.2.4.1.6.4.12 (p. 20). "sustain" mean "survive"? What does this requirement imply?
- Transport one injured crewmember back to Earth in addition to normal crew complement (2.2.4.1.6.4.13) [Note: This implies an 8-crew module actually has a 9-crew capacity]
- "Autonomous", on-board crew training capability (2.2.4.1.6.5.1)
- Technical procedures & Ops data electronically available at point of execution (2.2.4.1.6.5.2)
- Single crewmember maintenance, normal and contingency operation (2.4.4.1.4.6.1)
- Modular systems; spares (2.2.4.1.6.6.2-3)
- IVA/EVA systems comply with NASA Std 3000 (MSIS) (2.2.4.1.6.6.6)
- Prox Ops require direct operator viewing;
  areas requiring EVA access are viewable direct or by TV (2.2.4.1.6.6.4-5)
- Piloted rovers are pressurized if range ≥ 10 km (2.2.4.1.6.6.7); Crew ≥ 2 if range ≥ 1 km (2.2.4.1.6.8.4)
- EVA outside time <8 hrs; two crew minimum per EVA; all crew have personal suits (2.2.4.1.6.8)
- Propellant autonomous transfer in LEO, but astronaut backup of <40 hrs EVA, <100 hrs IVA (2.4.4.8.3.1)
- Rescue using ETO to LSurf (2.2.4.8.2.1.4, -2.2.5)
- Communications (2.2.4.1.7, including Table 2.2.4.1.7.2-1)
- Surface
  Base is on equator at Mare Tranquillitatis (24° E); farside observatory at 141° E on equator (2.2.4.5.1)
  Mass allocations are 105 t, 260 t, and 280 t for Outpost, Experimental, and Operational phases (2.2.4.5.3)
  Base will service, maintain, and store all launch vehicles (2.2.4.5.6)

Deviations from SRD:

Dual failure tolerant subsystems, operable from redundant locations (2.2.4.1.6.4.3)

TIA will assume one engine out for cryopropellant-based propulsion and excludes this requirement for storable bipropellant-based systems.

Lunar LOX Utilization

Greatest payoff is determined to utilize LLOX just for LSurf <—> LLO and return-to-Earth from LLO. Amount of LH2 to be carried is just sufficient to provide these capabilities.
No net return of LLOX to LEO.

Derived Requirements:

- 1.2 t radiation shielding per 4 crew (for LPV only) (2.2 t for 8 crew cab)
- Venting of modules and EVA as one of the two escape paths is permissible (private communication with D. Bland, 2-22-89; see also 2.4.4.1.6.4.6)
- Optimize O/H mixture ratio against IMLEO for use of LLOX
- LCSV is only partially-loaded when used for crew 4
- All landers are Lunar surface-based [early landers expendable?]

MARS EVOLUTION

Extracted from signed SRD dated March 3, 1989, Section 2.3 Mars Evolution Case Study (pp. 49-98)

Groundrules:

- Establish a Martian moon "gateway", followed by a Mars surface facility (2.3.1.A, C)
- Significant science research capability (2.3.1.B)
  Explore *both* Martian moons (2.3.4.1.6.1.A)
- Reusable transportation system (2.3.4.2.2)
  But aerobrake for A/C only at Mars for cargo (2.3.4.2.3.4.2) (implies expendable)
  Expendable ECCV for contingency direct entry (2.3.4.2.2)
- Commonality: Minimum number of vehicles (2.3.4.2.1.2)
- All-up, split, or convoy missions allowed (2.3.4.1.7.A)
  Landings shall be in daylight (2.3.4.1.7.D, 2.3.4.2.4.4.2.A.i, 5.4.2.A.i)
- Space Station Freedom support (2.3.4.3)
  5, 6, and 7 crew (phased); 10 t cargo; 15 month vehicle processing time
- LEO Node (2.3.4.4)
  Provides a free-flyer spacecraft for man-tended LEO assembly/checkout, including:
  Mating/assembly, construction, deployment/retrieval, on-orbit checkout, debris protection.
- Technology Development (2.3.4.9): Technology Level 6 for MAb and EAb by 1996, NEP by 2005, NTR by 2007

Requirements:

- Human mission departs Earth in 2004 (2.3.4.1.2)
  Crew ≥ 3. Two are IVA/EVA proficient, two are EMT (2.3.4.1.8.1)
- Development Phases for Martian surface base (2.3.4.1.2)
  Science outpost: instrumentation
  Human-tended: 5 crew, 1-yr TOD. Not permanently occupied.
  Operational: 7 crew, 2-yr TOD. Global access to Mars. Use indigenous resources for life support.
- H/O propellant production at the gateway (2.3.4.1.4)
- Tethers: Facility for momentum exchange and propellant transfer at the gateway.
  Determine applications and assess advantages. (2.3.2.F, 2.3.4.1.4, 2.3.4.5.2.2, 2.3.5.2.E)
- Personnel Transportation
  LEO <—> Gateway

Accommodate 5 crew, mixed gender. Increase to 7 crew in 2014 (2.3.4.2.2.2)
Accommodate a Mars descent/ascent vehicle (5-crew capacity) plus 20 t add'l cargo (2.3.4.2.2.3)
Gateway <—> MSurf
Land to elevations up to 15 km (2.3.4.2.4).
5 crew, mixed gender, increased to 7 crew in 2014 (2.3.4.2.4.2)
Include 10 t cargo, Gateway —> MSurf (increase to 25 t if no ascent cab included) (2.3.4.2.4.3)

- Cargo Transportation (2.3.4.2.3.3)
  LEO —> Gateway: 150 t equipment , enhanced to TBD in 2014.
  Gateway —> MSurf: 50 t equipment (of the 150 t above), to TBD in 2014. (2.3.4.2.5.3, -2.3.3)
- 1252 d flight time design (for Mars flyby abort); up to 400 d at Mars (2.3.4.2.2.1)
- Artificial-g spaceship, ≥1/3 gee, ≤4 rpm (2.3.4.2.2.4.3, but not on ascent/descent vehicles, 2.3.4.2.4.4.3)
  Determine operational limits of coriolis forces (2.3.4.6.2.4)
- Aerocapture at Mars and Earth for personnel vehicle (2.3.4.2.2.4.2), at Mars for cargo (2.3.4.2.3.4.2)
  No restriction on aerobrake L/D ratio.
  Mars entry velocity ≤9.5 km/s (2.3.4.1.7.E). (No limit on max-deceleration)
  Earth entry velocity ≤13.5 km/s (2.3.4.1.7.G). (No limit on max-deceleration)
  Mars Descent Vehicle includes aerobrake to lower apoapsis to circularize for lighting control (2.3.4.2.4.4.2, -5.4.2)
- Transportation vehicles shall be sized for pre-set ΔV's and durations (2.3.4.2.2.1, -3.1, and -4.1)
  Requirements change after 2014 for personnel carrier (2.3.4.2.2.1.B; to accommodate NTR, 2.3.4.2.2.4.1.B)
- Chemical propulsion initially for both cargo and human transportation (2.3.4.2.2.4.1, -3.4.1)
  Nuclear Thermal Rocket (NTR) for MPV in 2014 (2.3.4.2.2.4.1.B)
  Nuclear Electric Propulsion (NEP) for MCV in 2014 (2.3.4.2.3.4.1.B)
  All ascent/descent vehicles remain chemical propulsion at all times (2.3.4.2.4.4.1, -5.4.1)
- Multi-impulse TMI and TEI are permitted (2.3.4.1.7.B, -.F) (to minimize gravity and plane change losses)
- No single-point failure in subsystems of safety-critical systems (2.3.4.1.8.4.3)
  Dual failure tolerant subsystems, operable from redundant locations (2.3.4.1.8.4.3)
  [Note: TIA assumes one engine out for cryopropellant-based propulsion and excludes this requirement for storable bipropellant-based systems]
- Emergency operation of all systems by EVA-suited crewmember (2.3.4.1.8.4.10)
- Protect against excessive cosmic and solar radiation.
  Capability accessible within 30 minutes (2.3.4.1.8.4.2)
  [Note: No protection against cosmic radiation will be provided by TIA]
  Radiation protection on MPV: provide 5 g/cm2 shielding (2.3.4.2.2.4.4)
  [Assumption: does not include slant-path or astronaut mutual shielding benefits]
  [Assumption: provided only in a radiation storm shelter, not for entire hab module]
- Crew work/rest scheduling
  6 work days/wk @ 8 hrs duty time/work day, 2 hrs/day exercise (2.3.4.1.3.2.2,-.3)
  Until permanent human presence is established, all crew have same day off (2.3.4.1.8.2.4)
- All hazardous materials stored outside of pressurized elements (2.3.4.1.8.4.4)

- Isolation and rapid egress from any habitable element in emergency (2.3.4.1.8.4.5)
  All habitable elements shall have redundant escape paths (2.3.4.1.8.4.6)
  [Note: but not necessarily rapid. See use of EVA escape path, under Derived requirements]
- Pressure integrity checks; adequate day/night lighting; pressure hatches (2.3.4.1.8.4.7-9)
- "Autonomous", on-board crew training capability (2.3.4.1.8.5.1)
- Single crewmember maintenance, normal and contingency operation (2.3.4.1.8.6.1)
  Modular systems; spares (2.3.4.1.8.6.2-3)
  Technical procedures & Ops data electronically available at point of execution (2.3.4.1.8.5.2)
- IVA/EVA systems comply with NASA Std 3000 (MSIS) (2.3.4.1.8.6.6)
  Propellant transfer ops require ≤100 p-hr IVA, ≤ p-hr EVA (2.3.4.8.3.1)
- Prox Ops require direct operator viewing;
  areas requiring EVA access are viewable direct or by TV (2.3.4.1.8.6.4-5)
- EVA outside time <8 hrs; two crew minimum per EVA; all crew have personal suits (2.3.4.1.8.8)
  All rover excursions ≥ 1 km require 2 crew (2.3.4.1.8.8.4). ≥ 10 km require pressurized rover (2.3.4.1.8.6.7)
- Communications (2.3.4.1.9.2, including Table 2.3.4.1.9.2-1)
  10 Mbps MTE, 20 Mbps ETM at 2.5 AU
  [Note: 10 Mbps MTE exceeds estimated need]
- User requirements (Instrument Packages): Solar; Cosmic Dust; Cosmic Rays; Gamma Bursts; Biomedical. Engineering characteristics provided in Study Data Book (2.3.4.2.2.5).
- ETO Cargo (2.3.4.1.1): Launches ≥45 day intervals, 4 launches/yr.
  140 t to i=28.5°, 500 km LEO. Accommodates 12.5 m diameter x 25 m long cargo load.
  570 t/yr to 500 km. Dry to LEO ≤180 t per two *consecutive* years (2.3.4.1.1)
  ETO Crew (2.3.4.8.2.2.3): Mission crew of 5 to Freedom, plus 5 t cargo; servicing crews to 6.
- Planet Surface System Requirements: Section 2.3.4.5.
  Landing site is Chryse basin complex (equator, 33.5° W).
  Phobos/Deimos Surface Base mass allocation: 100 t (2.3.4.5.2) (for PhLOX, PhLH2 production)
  Mars Base mass allocations: Outpost, 35 t; Human-tended, 120 t; Operational, 150 t (2.3.4.5.1.3)
  ISRU for Mars H2O, MLOX, MLH2, construction. (2.3.4.5.1.5)
  Provide surface transportation, including shielding against GCR and SPE (2.3.4.5.1.7)
  Mars Target Science:
  Geology, Geophysics, Atmospheric, Particles and Fields, Exobiology, Resource Assessment
  Mars Platform Science (Laboratory): Geochem/Petrology/Paleomag, Biochemistry, Life Sciences
  Mars Surface Science: Sample return, mobile vehicle, sampling, automated geophys/atmo stations

Deviations:

- Dual failure tolerant subsystems, operable from redundant locations (2.3.4.1.8.4.3)
  TIA will assume one engine out for cryopropellant-based propulsion and excludes this requirement for storable bipropellant-based systems
- MCV is Expendable
- MCSV is Reusable (post Gateway Operational phase)
- TMIS stages expended (make up part of MPV and MCV)
- ΔV of 200 m/s post-EOC not included.
  STV assumed to accomplish this

Derived Requirements:

- Provide windows (to view space and Mars surface non-electronically, 2.3.4.1.8.3.2)
- Venting of modules and EVA as one of two escape paths is permissible
  (private communication with D. Bland, 2-22-89; see also 2.3.4.1.8.4.6)
- Vehicles must operate autonomously during non-critical periods
  [48 work-hrs/week, leaving 120 hrs/week (71% of time) when no crew are on duty (2.3.4.1.8.2)]
- Dual habitation modules required on MPV and on MSurf

MARS EXPEDITION

Extracted from signed SRD dated March 3, 1989, Section 2.4 Mars Expedition Case Study (pp. 99-127)

Groundrules:

- Achieve a human landing on Mars ASAP ("earliest feasible mission opportunity", 2.4.2.B, 2.4.3.1.A)
- One human mission only (2.4.4.1.3)
- All space transfer vehicles are expendable (2.4.4.2.2.,-3)

- Transportation system to be launched *intact*, with <u>no</u> assembly in LEO (2.4.4.2.2.4.5, -3.4.3; 2.4.3.1.B)
  But both support propellant transfer in LEO (2.4.4.2.2.4.5, -3.4.3)
- No orbital nodes are required (2.4.4). Space Station Freedom provides LSS qualification,
  but no unique capabilities or accommodations (2.4.4.3)
- 1995 technology (2.4.4.1.3) (but "allowing for very high leverage technology extensions", 2.4.4.1.3)
  Technology Level 6 for MAb by 1994, ECCV brake by '96, Mars landing Nav and Hazard Avoidance by '95 (2.4.4.9)
- Maximum use of orbiters and landing beacons from precursor mission(s) (2.4.4.1.3)
- Aggressive Phase C/D schedules (4-5 yrs) (2.4.4.1.2)

Requirements:

- Split/sprint trajectory (2.4.4.1.1.A), with free return abort for piloted vehicles (-.C)
- Human transportation vehicle is operational by July 2002 (2.4.4.2.2)
  Vehicle sized for a single mission only (2.4.4.1.3)
  Crew of 3 (2.4.4.2.2.2, 2.4.3.1.E); Two are IVA/EVA proficient, two are EMT (2.4.4.1.4.1)
  No cargo capacity on MPV (2.4.4.2.2.3)
- Cargo transportation vehicle is operational by March 2001 (2.4.4.2.3)
  Cargo capacity is the MDV, plus 10 t additional equipment with TBD dimensions (2.4.4.2.3.3, -4.3)
- Zero-g spaceship (2.4.4.2.2.4.3)
- Mars aerocapture (2.4.3.1.H). [Note: Differs in this respect from CS-1.0 of FY88]
  Aerobrakes of L/D between 0.9 and 1.2 for both cargo and piloted vehicles (2.4.4.2.2.4.2, -3.4.2)
  Mars entry velocity ≤9.5 km/s (2.4.4.1.1.E). Max-deceleration ≤5 gee (-.H)
- Direct entry at Earth (2.4.3.1.H).
  <u>Note</u>: Not a *requirement*. This appears only in 2.4.3, Ref. Mission. TIA accepts Direct Entry as baseline, however.
  Earth entry velocity ≤16.0 km/s (2.4.4.1.1.G). Max-deceleration ≤5 gee (-.H)
- Transportation vehicles shall be sized for pre-set ΔV's and durations (2.4.4.2.2.1, -3.1, and -4.1)

- 730.3 d flight time design; 30 d at Mars (2.4.4.2.2.1) [Note: 730 d is much greater than "sprint" times]
- Chemical propulsion for both cargo and human transportation (2.4.4.2.2.4.1, -3.4.1)
- Multi-impulse TMI and TEI are permitted (2.4.4.1.1.C, -.F) (to minimize gravity and plane change losses)
- MDV uses storable propellants; aerobraking; no rad protection (2.4.4.2.4.4.1-4)
  MDV can land at altitudes to +5 km (2.4.4.2.4). Landing must occur in daylight (2.4.4.1.1.D)
  All 3 crewmembers to Mars surface for 20 days (2.4.4.2.3.3, -4.2)
  10 t cargo with TBD dimensions (2.4.4.2.4.3)
    Note: -2.4 reads "transfer crew and cargo from ... orbit to ... surface and back to ... orbit"
    TIA assumption is that transportation of 10 t cargo from surface back to orbit is *not* required
- No single-point failure in subsystems of safety-critical systems (2.4.4.1.4.4.3)
  Dual failure tolerant subsystems, operable from redundant locations (2.4.4.1.4.4.3)
    [Note: TIA assumes one engine out for cryopropellant-based propulsion and excludes this requirement for storable bipropellant-based systems]
- Emergency operation of all systems by EVA-suited crewmember (2.4.4.1.4.4.10)
- Protect against excessive cosmic and solar radiation.
  Capability accessible within 30 minutes (2.4.4.1.4.4.2)
    [Note: No protection against cosmic radiation will be provided by TIA]
  Radiation protection on MPV: provide 5 g/cm2 shielding (2.4.4.2.2.4.4)
    [Assumption: does not include slant-path or astronaut mutual shielding benefits]
    [Assumption: provided only in a radiation storm shelter, not for entire hab module]
- Crew work/rest scheduling
  6 duty days/wk @ 8 hrs duty time/work day, 2 hrs/day exercise (2.4.4.1.4.2)
- All hazardous materials stored outside of pressurized elements (2.4.4.1.4.4.4)
- Isolation and rapid egress from any habitable element in emergency (2.4.4.1.4.4.5)
  All habitable elements shall have redundant escape paths (2.4.4.1.4.4.6)
    [Note: but not necessarily rapid. See use of EVA escape path, under Derived requirements]
- Pressure integrity checks; adequate day/night lighting; pressure hatches (2.4.4.1.4.4.7-9)
- "Autonomous", on-board crew training capability (2.4.4.1.4.5.1)
- Single crewmember maintenance, normal and contingency operation (2.4.4.1.4.6.1)
  Modular systems; spares (2.4.4.1.4.6.2-3)
  Technical procedures & Ops data electronically available at point of execution (2.4.4.1.4.5.2)
- IVA/EVA systems comply with NASA Std 3000 (MSIS) (2.4.4.1.4.6.6)
  Propellant transfer ops require ≤100 p-hr IVA, ≤ p-hr EVA (2.4.4.8.3.1)
- Prox Ops require direct operator viewing;
  areas requiring EVA access are viewable direct or by TV (2.4.4.1.4.6.4-5)
- EVA outside time <6 hrs; two crew minimum per EVA; all crew have personal suits (2.4.4.1.4.8)
- Communications (2.4.4.1.5, including Table 2.4.4.1.5.2-1)
  10 Mbps MTE, 20 Mbps ETM at 2.5 AU. "Continuous communication is *not* required"
  [Note: 10 Mbps MTE exceeds estimated need. Range of MPV-to-Earth never exceeds 1.8 AU]
- User requirements (Instrument Packages): Solar; Cosmic Dust; Cosmic Rays; Biomedical.
  Engineering characteristics provided in Study Data Book (2.4.4.2.2.5). Minimal science equipment (2.4.3.1.F)

- ETO Cargo (2.4.4.8): Launches ≥45 day intervals, 4 launches/yr.
  140 t to i=28.5°, 500 km LEO. Accommodates 12.5 m diameter x 25 m long cargo load.
  Supports first MCV mission of Jan 2001.
  ETO Crew (2.4.4.8): 2 launches/yr. Not more than 3 days after a cargo launch.
- Planet Surface System Requirements: Section 2.4.4.5.
  Landing site is Ganges Chasma (8.3° S, 44.2° W).
  No ISRU. Five 6-hr EVAs per crew member. *No* personnel rovers.
  Geology, Geophysics, Atmospheric, Exobiology experiments

Deviations from SRD:

Dual failure tolerant subsystems, operable from redundant locations (2.4.4.1.4.4.3)
TIA will assume one engine out for cryopropellant-based propulsion and excludes this requirement for storable bipropellant-based systems. TEIS provides 3-engine out, however.

Mars and Earth entry velocity max-deceleration ≤5 gee (2.4.4.1.1.H)
No established requirement for ≤5 gee. Suitably restrained and unconditioned crewmembers may take 10.5 gee for 1 minute if oriented in the +Gx direction (Ref.: NASA Std 3000, Fig. 5.3.3.1.-1)]

Communications links of 10 Mbps MTE, 20 Mbps ETM at 2.5 AU
(2.4.4.1.5, including Table 2.4.4.1.5.2-1)
TIA will size for 10 Mbps (which exceeds estimated continuous needs), but at 1.4 AU max)

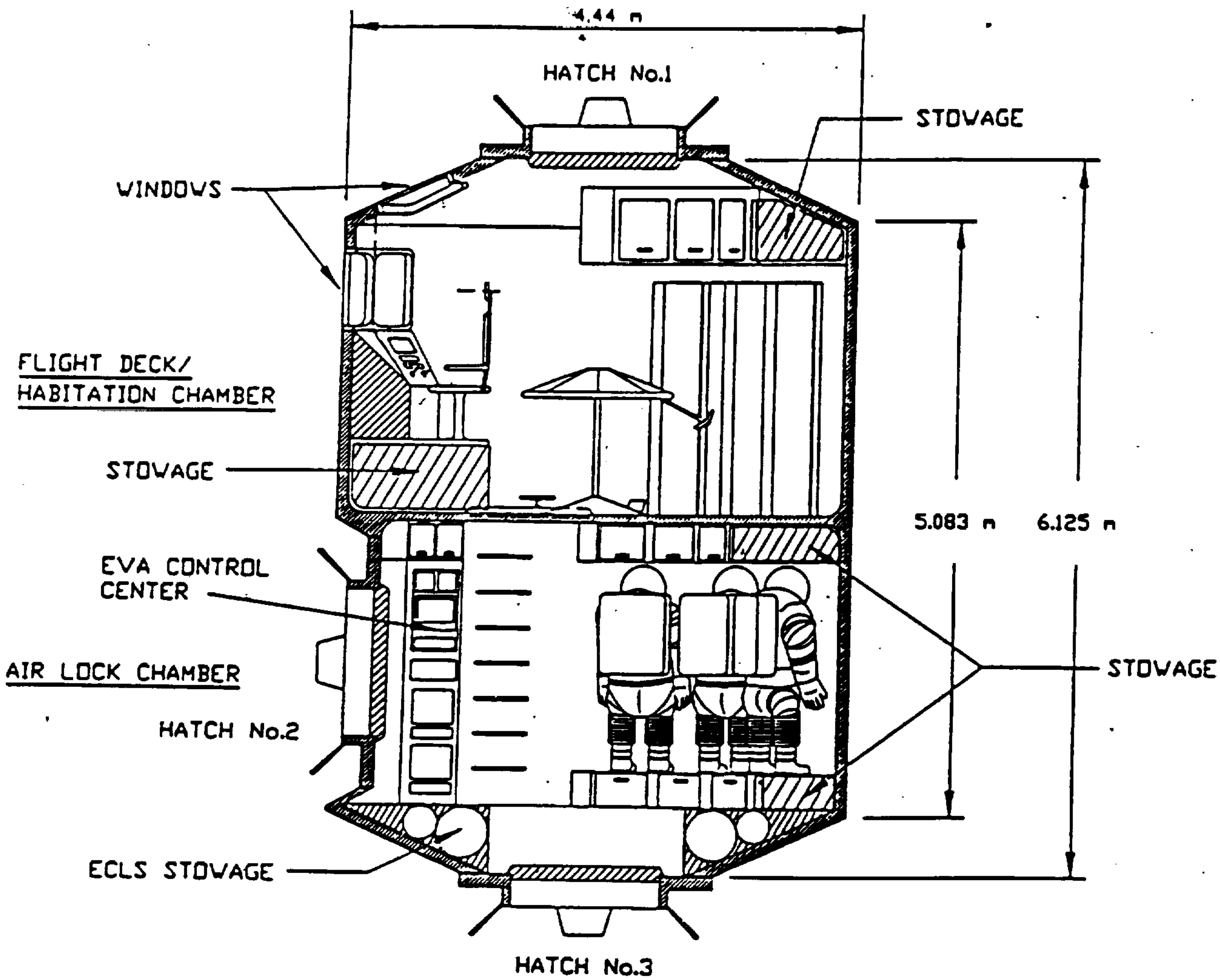
Protect against excessive cosmic and solar radiation.
Radiation protection on MPV: provide 5 g/cm2 shielding (2.4.4.2.2.4.4)
No added protection against *cosmic* radiation will be provided by TIA
The amount of 5 g/cm2 shielding is unnecessarily inadequate for Solar Particle Events

ETO Shroud 25m in length (2.4.4.8.2.1.3)
TIA recommends elimination of shroud for launch of MCV and MPV hardware (dry).
Substitute Mars Aerobrake (MAb) for shroud. MAb is 27.1 m in length

Derived Requirements:

- Provide windows (to view space and Mars surface non-electronically, 2.4.4.1.4.3.2)
- Venting of modules and EVA as one of the two escape paths is permissible (private communication with D. Bland, 2-22-89; see also 2.4.4.1.4.4.6)
- 48 duty-hrs/week, i.e., 24 hrs/week when no crew members are on-duty (from 2.4.4.1.4.2)
  OR, if crews work together, 120 hrs/week (71% of time) when no crewmembers are on duty
- No recovery of ITV (hab elements, dry TEIS, etc.). No recovery of TMIS stages
- Access to landing sites up to ±10° latitude (inferred from 2.4.4.5.1 and 2.4.4.7.1)
- Mars Parking Orbit (MPO) is Low Mars Orbit (LMO) (e.g., near-circular at 300-500 km) (inferred from ΔV=4200 m/s for Ascent in para. 2.4.4.2.4.1)

By: Eagle Engineering (L. Guerra, B. Stump)



CREW TRANSFER MODULE SECTION

Figure C-9 2-Deck LCSV Habitat, Zero-Gravity

By: Eagle Engineering (L. Guerra, B. Stump)

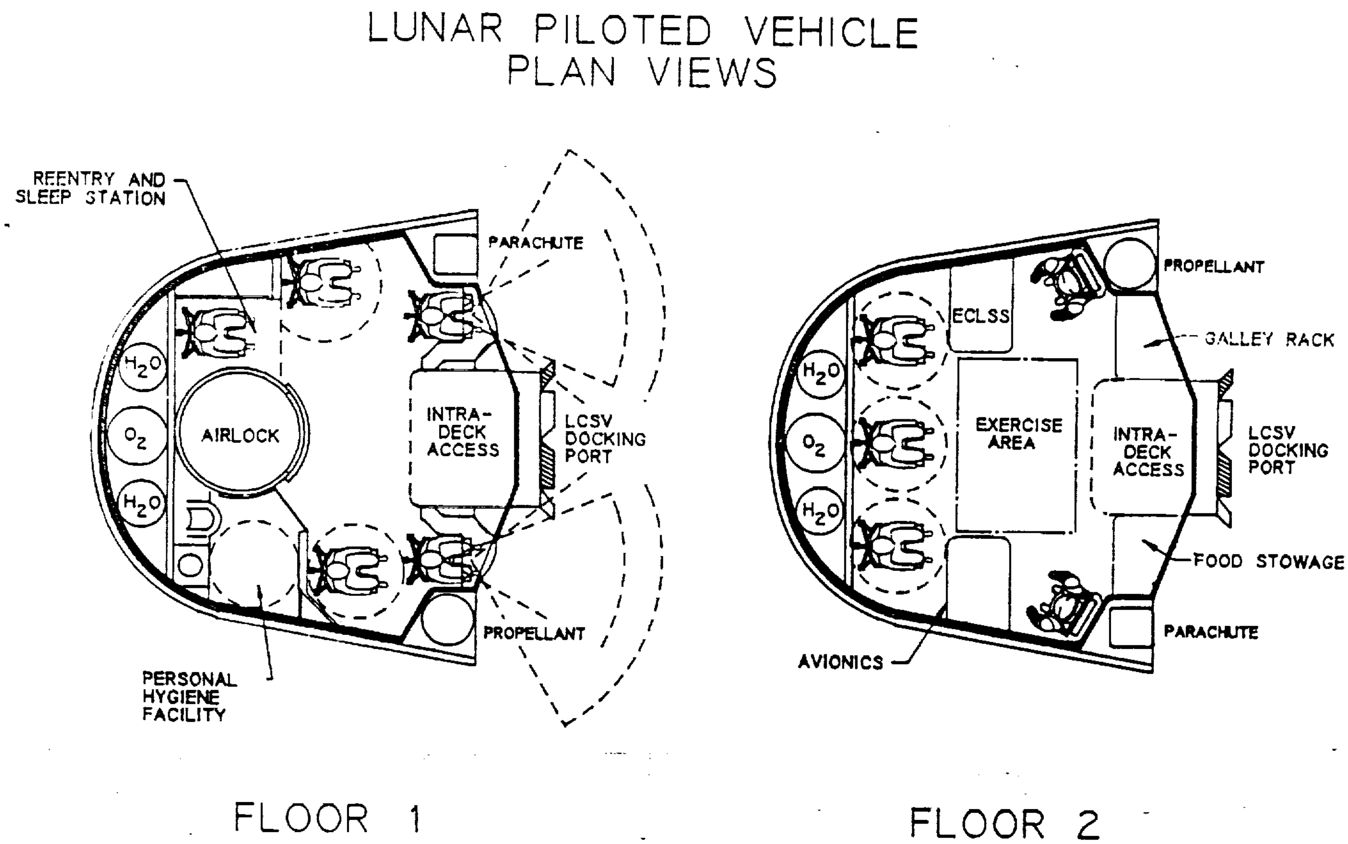


Figure C-10 Alternative LPV Habitat, Zero-Gravity

APPENDIX D. AEROASSIST ALTERNATIVES

Aeroassist is the use of aerodynamic braking in the atmosphere of a planet to reduce orbital energy. It may be applied to capture into a closed orbit from a hyperbolic encounter condition or for reduction of the size of an existing orbit. Its use in manned missions raises whole new issues in terms of man-rating requirements, but it does represent a technology that has a firm basis in the many years of entry maneuvering work performed on such programs as Gemini, Apollo, and Shuttle. At Mars, velocity reductions ranging from 2 to 6 km/s are required to capture, depending on encounter and captured park orbit conditions. For Earth capture, the delta-v's range from 1.2 to 8 km/sec with capture orbits varying from a low 1.5 hour period to highly elliptical 4 day orbit. For closed Earth orbits, the velocities vary from 2.4 km/s for GEO return to 3 km/s for lunar return. At the low end of aero energy reduction, GEO return, an aeroassist device is performance effective if its mass fraction is less than 15% of the captured payload weight. At the higher end of the scale, sprint class Mars missions can have brake weights exceeding the mass of the payload itself and still result in IMLEO far less than an all-propulsive approach (Figure D-1). Packaging considerations show preference for low L/D blunt aerobrake concepts, while higher L/D biconic shapes are attractive for g reduction in the fast encounter regimes. Manned mission aerobrakes are generally large in size (ranging from 14 m in diameter for a lunar return brake to 40 m for a Mars mission device) and thus the method of on-orbit assembly is a major concern. The primary technological areas to be considered are aerothermal, thermal protective system (TPS), guidance, navigation, & control (GN&C), on-orbit assembly techniques, and atmospheric characterization.

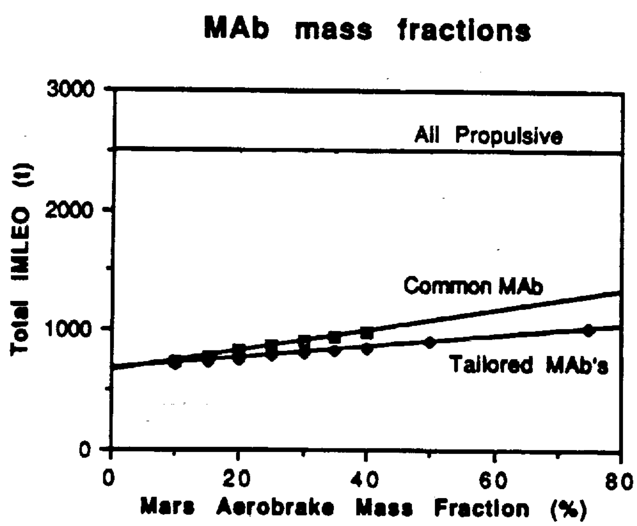


Figure D-1 Aerobrake Mass Ratio Sensitivity

The Mars Rover Sample Return represents the next major mission to the planet Mars. It is anticipated that aeroassist will play a major role in the mission with automated capture phases at both Mars and the Earth. Manned missions to Mars are hoped to be accomplished early in the next century as a major new chapter in the exploration of space by humans. The OEXP cycle 2 case studies investigated several options for manned missions. Various vehicle configurations were studied which utilized aerocapture at Mars and the Earth. Abort considerations as well as man-rating in general will be a strong driver for the design of aerocapture maneuvers for such missions.

AEROTHERMAL

Aerothermal characterization of the entry environment is crucial to correctly designing the aerobrake's TPS. Previous entry programs had extreme amounts of conservatism built into their entry heat shields because of a lack of knowledge of the thermal environment. In many cases this level of conservatism will result in marginal performance of an aerobrake. One of the biggest areas of uncertainty currently is the contribution of non-equilibrium heating. Particularly for high lift configurations flying near the skipout boundary, this can be a significant heating contribution. Currently the Aeroassist Flight Experiment (AFE) is tasked with obtaining flight data in this area for a GEO return mission. The use of computational fluid dynamics (CFD) codes should eventually reduce much of the uncertainty in characterization of the thermal environment, though there appears to be a great deal of disagreement as to how much. The impact of real-gas effects and CO2 dissociation, while significant, does not appear to be a first order driver.

TPS

The development of advanced high temperature TPS is important for the thermal regimes as well as for aerobrake design flexibility. The use of elliptical intermediate park orbits (Figure D-2) and exo-atmospheric deceleration burns can reduce the entry energies that must be dissipated. Very high entry speeds at Earth (in excess of 13 km/s) will demand the use of ablator technology. Most high temperature ablators are inherently heavy, however, which makes the investigation of lightweight ablator technology important. Other problems inherent with ablators are their outgassing deposition onto sensitive optical/thermal surfaces and questions of multiple use because of the altered aerodynamic surface and reflectance properties. Medium temperature TPS options include derivative Shuttle tiles. Although these materials are fairly lightweight they are extremely fragile and may require new bonding techniques for extended exposure to the space environment. Multilayer metal foil or advanced carbon/carbon materials would represent more durable TPS options. In the low end of the temperature spectrum, flexible ceramic TPS such as the NASA/ARC-developed TABI (Figure D-3) can allow large diameter lightweight aerobrakes. These concepts can be automatically deployed on orbit, which reduces the assembly problem. The very significant issues of embrittlement and dynamic flutter must be investigated thoroughly, however, before these concepts can be utilized.

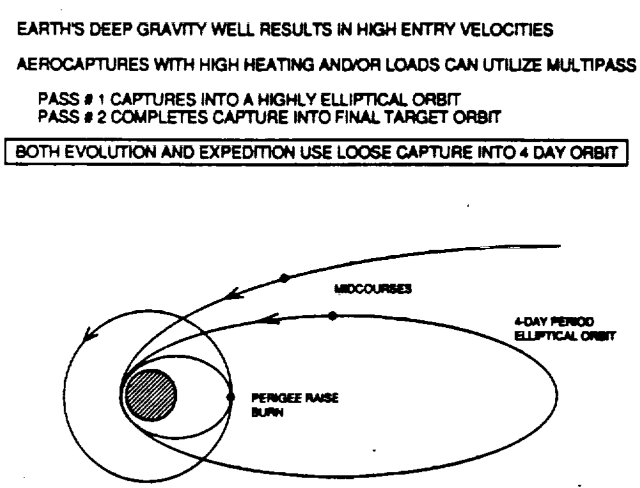


Figure D-2 Earth Aerocapture for G-Load Relief

GUIDANCE, NAVIGATION, AND CONTROL (GN&C)

The area of GN&C is critical to maintaining control of the vehicle through the atmospheric flight phase. Encounter navigation is a driver to the feasibility of the aeroassist maneuver. Uncertainty in the entry location results in rapid increases in the basic loading to the vehicle as well as rapid increases in the propagation of errors to the exit state. These errors cannot simply be "flown out" through the use of greater amounts of lift. The requirement for man-rating may maximize the use of stand-alone concepts rather than those requiring outside infrastructure. Two basic options are possible for accurate navigation state determination. The simplest is the use of radionavigation to an existing NavSat in orbit around the encountered planet. This obviously requires the development of infrastructure. Results of a Mars NavSat study are shown in Figure D-4. Very good accuracies are achieved with even late acquisition of signal. The only major technical issue involves the acquisition of navigation signals at very long ranges from the planet. This approach does levy a significant infrastructure requirement. In the case of the Earth, this infrastructure will exist in the early-1990's with the completion of the Global Positioning System (GPS) satellite network.

In the case of Mars a system of at least two satellites (for redundancy) would have to be deployed. Development of such a Mars infrastructure is more likely by the time of a manned Mars mission; its existence would be more problematical at the time of MRSR missions.

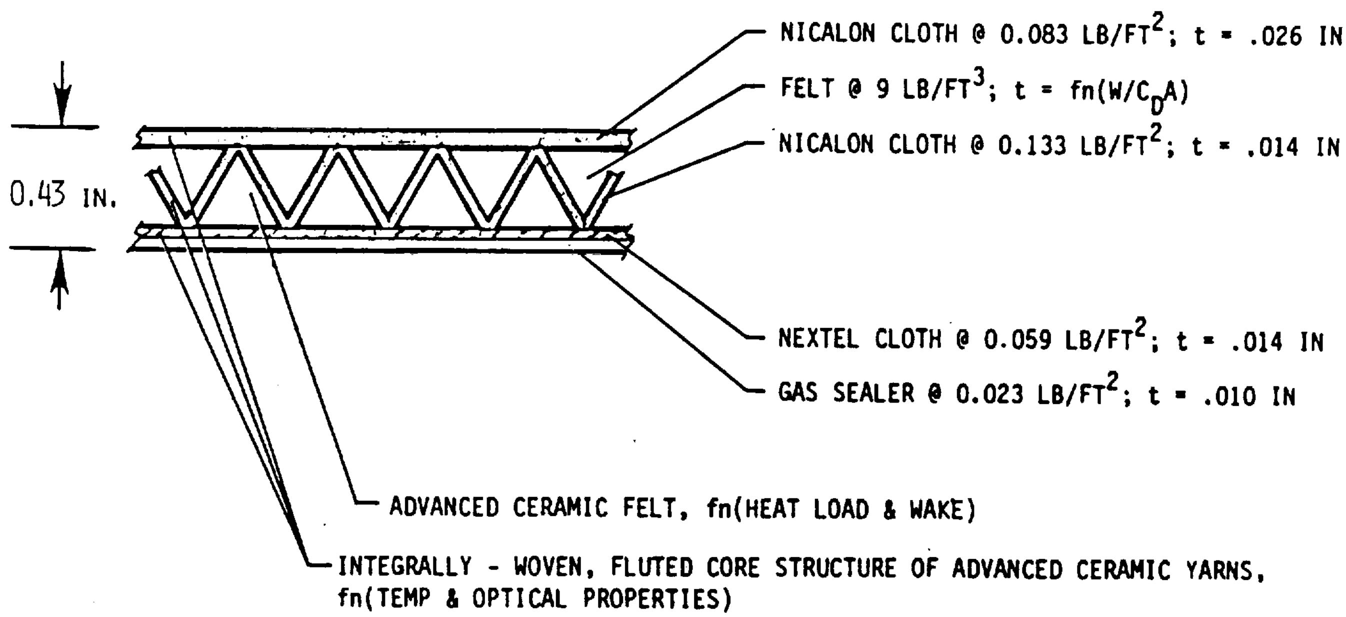


Figure D-3 Tailorable Advanced Blanket Insulation

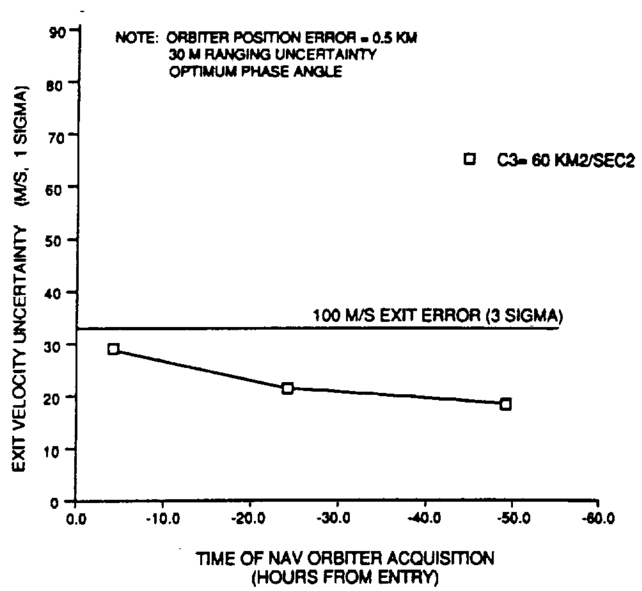


Figure D-4 Naviagation Accuracy from Cooperative Orbiter

The other navigation technique is the use of onboard optical measurements close in to the planet. This form of navigation was used by Apollo as a backup to ground-based measurements. Good accuracies are achieved with reasonable instruments if the navigation process can proceed to within a few hours of entry (this cutoff time depends on the energy of the encounter orbit). Figure D-5 shows study results using 1 arc second resolution angle measurements and a 2 km Deimos position uncertainty. Sextant type hardware with these levels of accuracy have been built and tested previously. Moon ephemeris accuracies of this order can be obtained by on-board estimation techniques or by an orbiting spacecraft's in-situ observations (perhaps even by an Earth-based Space Telescope?). Because of a lesser reliance on existing infrastructure this type of navigation might be more attractive than NavSat radionavigation for an initial unmanned mission such as MRSR. On the other hand, the autonomous requirement of long-distance unmanned vehicles means that the validation of recognition and sensing techniques for such an approach are especially demanding. Additionally, in the case of manned missions, implementation of on-board optical techniques may be required to improve man-rating by reducing dependency on external systems.

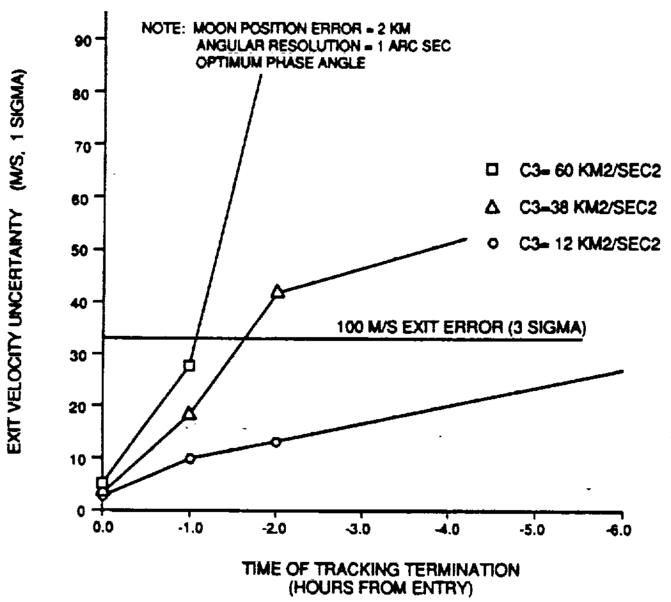


Figure D-5 Optical Navigation Measurements Using Deimos

Descent to landing presents a challenge to the navigation system in reaching a precision landing site. A variety of techniques are possible including radio ranging to an orbiting NavSat and/or ground beacon, landmark tracking, or correlation of radar terrain profiles. All the above options can provide sub-kilometer accuracy at retro-ignition to a desired landing site. The radio ranging options require the most infrastructure at the planet but are the simplest for the vehicle system to implement. Landmark tracking is the most robust stand-alone system but is also the most hardware/software intensive option for the vehicle. In addition the impact of shock refractions could make it unusable in the hypersonic phases of entry. The radar correlation option is simpler to implement and can make use of existing landing radar but does suffer from potential ambiguities in its results.

Aeroassist guidance must provide accurate end conditions while maintaining g-load and heating constraints. Low control rates are desirable both for minimum fuel consumption as well as from a crew disorientation standpoint. Adaptive guidance techniques that are responsive to changing environmental and vehicle conditions appear to be necessary. Robust algorithms that minimize extra control modes will result in a safer, simpler system overall. In the work done to date, the use of roll control of the vehicle lift vector alone is adequate to control the entry profile with acceptable exit errors. The use of atmospheric grazing passes with lift predominantly down can provide load relief for high energy missions, but TPS requirements rise due to the longer heat soak times.

The control of the vehicle in the aeropass should minimize the number of required systems in order to simplify its design. It is fairly clear at this point that the most efficient method of trajectory control is through the use of vehicle lift. Schemes for direct variation of ballistic coefficient alone are very marginal to entry dispersions. Simple rate damping and roll control of attitude by RCS jets would be a desirable goal since it provides the simplest implementation. The onboard guidance system should be able to handle a certain amount of uncertainty in the vehicle angle of attack but this cannot be excessive for heating as well as control reasons. The vehicle angle of attack is impacted by uncertainties in the aerodynamic properties as well by shifts in its center-of-gravity (cg) location. The aerodynamic properties can be determined by a combination of wind tunnel and CFD testing, probably to within a resulting ±1° angle of attack. If it is assumed that the net angle of attack is desired to be within ±2°, then the cg tracking's contribution must be maintained within ±1°. Figure D-6 shows vehicle envelopes for a blunt cone's cg with this constraint. Although the cone is relatively insensitive to longitudinal cg shifts, the lateral location requirements are quite tight (11 cm for a 39 m diameter aerobrake). Analogous, but inverse oriented results are obtained if a biconic entry vehicle is used, Figure D-7.

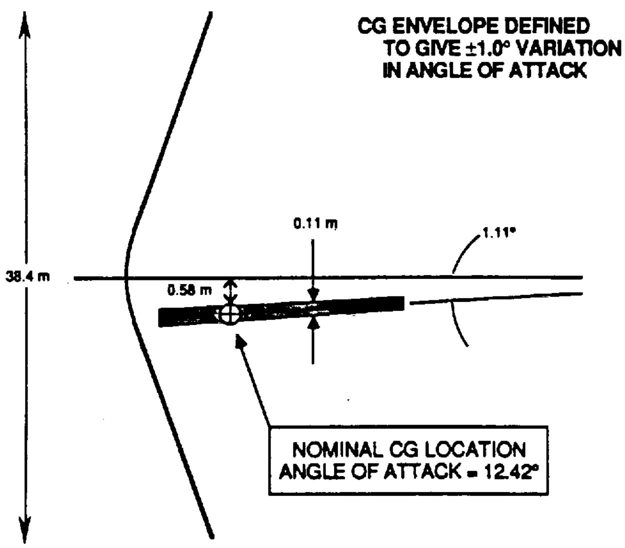


Figure D-6 Low L/D Aerobrake–cg Requirements

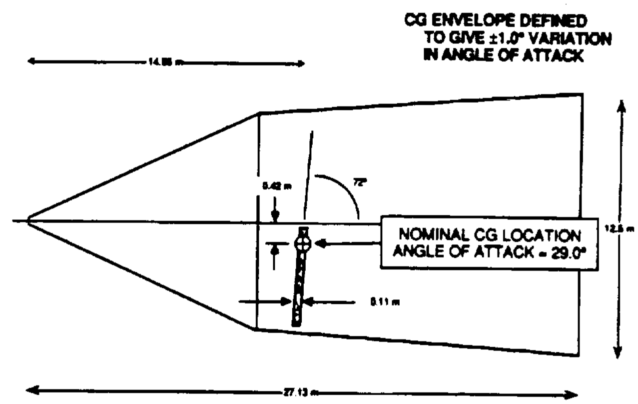


Figure D-7 High L/D Aerobrake–cg Requirements

Because manned vehicles must fly in deep space for extended periods of time with crew motion, consumables relocation and main propellant boiloff, the centering of the vehicle cg at the time of entry will be a significant issue. Relocation of personnel and habitat items to predetermined locations prior to entry will be required as a first step. Stowage of antennae and any other low-g equipment will have to be accounted for to first order in the overall cg design of the vehicle. The control of propellant location will be a significant a design driver requiring the use of propellant baffles and/or traps perhaps combined with settling burns prior to entry. Propellant cg control within tankage is a significant issue for any propulsive vehicle but design solutions do exist. The use of calibration roll maneuvers can be undertaken which measure the cg location via the use of strategically placed accelerometers. The sensitivity of these instruments is not unreasonable for vehicle roll rates in the 10-20 deg/s range. Pre-entry correction of the measured cg location can then be accomplished by shifting movable mass. Finally, the vehicle attitude can be actively controlled in the aeroassist phase, either through the use of flaps or by the actuation of a control mass (inert or propellant). This last option is clearly the most mechanism intensive but may be required for some configurations.

AEROBRAKE ASSEMBLY ON-ORBIT

For near-term unmanned missions such as MRSR the current emphasis is to utilize aerobrake and packaging concepts that do not require any on-orbit deployment or assembly. This is made possible by the large payload diameter capability of the Titan IV and Space Shuttle launch vehicles as well as the relatively small size of the mission spacecraft. However, because of the large size of manned missions, their aerobrakes may require significant on-orbit assembly and preparation for flight. Various concepts have been developed in the course of OEXP and previous studies. The use of advanced launch vehicles with large (12.5 m) payload diameter capability would enable the use of large biconic aeroshells to be delivered to orbit intact. This solution was used in the OEXP Mars Expedition case study where the payload to be delivered to Mars orbit was reduced in size by the use of a separate unmanned cargo flight. Although this monolithic approach obviates the need for on-orbit assembly it does present a driver for launch vehicle evolution. Because of their obvious packaging flexibility, several concepts of blunt cones were investigated as well. The use of flexible ceramic TPS concepts enable deployable aerobrake concepts (so-called flex/fabric aerobrakes) which thus eliminates brake assembly (but not the subsequent outfitting of mission elements onto the aerobrake assembly). These aerobrakes could be remotely deployed with springs and latches after delivery to orbit similarly to furlable space antenna designs (Figure D-7). The critical technologies to enable these forms of space-deployable aerobrakes involves the development of high temperature flexible TPS as mentioned above. These concepts were utilized in the OEXP Mars Evolution and Lunar Evolution case studies.

Finally, a rigid space-assembled blunt aerobrake concept was developed as an alternative to the flex/fabric aerobrake used in the Mars Evolution case study. This concept made use of advanced carbon/carbon high temperature material with backing insulation to prevent re-radiation. Advanced carbon/carbon has very high thermal flux capabilities (around 100 BTU/ft2/s) as well as good structural capability so that it eliminates the need for separate structural & thermal systems on the front of the brake. This avoids tile bonding problems as well as allowing a minimum diameter because of its high thermal capability. The minimum thermal diameter for the Mars Evolution vehicle is about 70 ft. However, because this presents severe packaging problems a diameter of 100 ft. was used as a design point. This concept was then utilized as a reference case for on-orbit assembly.

In the case of space assembled aerobrakes the number of pieces as well as the complexity of their integration should be minimized. These goals are usually in conflict with each other. Above all, the concepts for assembly must be self-reinforcing under air load if at all possible to provide fail-safe features as well as reducing the complexity of the joint interface. Such a concept is shown in Figure D-8. Using the concept of a central core section surrounded by assembled petals its conical form tends to give compressive forces at all seam lines. The use of shear blocks mating into slotted guide receptacles can simplify the alignment process (Figure D-9). Flexible high temperature seals can be located in the recesses of grooves which will alleviate their thermal flux requirements. Because the joints are under compression with shear blocks taking up the majority of the interface load, the fasteners that hold the panels together can be minimized both in size and number. Depending on the level of infrastructure available at the on-orbit assembly facility these fasteners can either be engaged by an external device (e.g., a drill on the end of a remote manipulator) or by a device that remains with the brake (actuator motors permanently mounted into the petals themselves). The first approach would minimize aerobrake scar weight at the expense of facility complexity while the second would do the opposite.

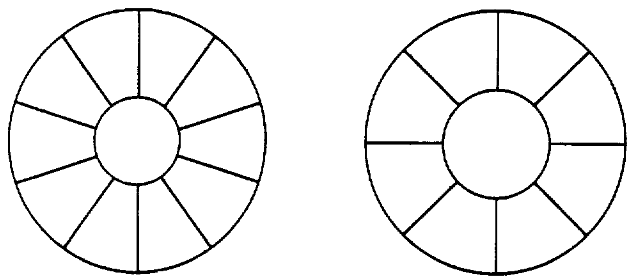


Figure D-8 Rigid Low L/D Aerobrake for On-orbit Assembly

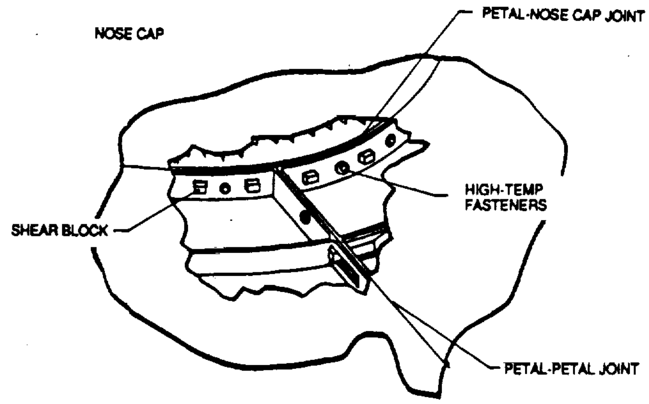


Figure D-9 Aerobrake Joining Concepts

The complexities of assembly of large structures on-orbit are significant and require extensive operations analysis. The problems associated with zero-g kinematics of large units apply both to human space walkers or teleoperated robots. The use of robotic and teleoperated assembly units must be maximized because of the great cost of EVA. Differential expansion due to severe solar heating can probably be minimized through the use of low expansion coefficient materials such as composites. Self checkout of structural integrity will result in significant instrumentation such as contact switches and strain gauges. Inspection of the finished structure will involve such techniques as differential laser interferometry to verify smoothness of fit. Finally, a low entry-velocity flight test into the Earth's atmosphere may be required as a final overall system check. The assembly operations for the Space Station Freedom will certainly drive solutions to many of these issues. A good technology base currently exists for high temperature joints and seals for the shuttle with its payload bay, landing gear and External Tank umbilical doors. However, this needs to be extended to higher heating rates.

ATMOSPHERIC UNCERTAINTIES

One of the first-order drivers to the design of the aerobrakes will be the degree to which the Mars atmosphere's uncertainties can be reduced. The bulk density variation of the atmosphere is a first order driver to the required L/D as well as the thermal margins because it alters the density altitude at which the vehicle must fly. Density shears and gravity waves present fluctuating atmospheric conditions which drive the control rates and exit errors of the vehicle. Table D-1 shows the impact of an unpredicted density dispersion upon the exit apoapsis accuracy for a representative manned Mars aerocapture. The small scale structure of these density phenomena (which affects dynamic loads) cannot be predicted but the larger scale structure (which impacts exit errors) should be, depending on the investment in in-situ observations. Dust storms alter the density structure of the atmosphere by solar heating of the optically thicker gas. However, much of this shift can probably be accommodated with far-encounter observations. The impact of winds will be felt most strongly in the entry to landing phase and its need for precision landing. Better characterization of the long-term behavior of the Martian atmosphere is called for with a dedicated orbiter that can make observations down to 20 to 40 km in altitude with good resolution (5 to 10 km).

Table D-1 Impact of Density Dispersion Exit Contditions

DISPERSION	EXIT ORBIT PERIAPSIS (KM)	EXIT ORBIT APOAPSIS (KM)	ΔV TO REACH PARK ORBIT (MPS)**	PEAK LOADS (g's)
NOMINAL	37.6	33845.4	12.3	6.91
LOW PRES ATMOS	41.0	36129.8	23.4	6.89
HIGH PRES ATMOS	34.9	31323.2	28.1	6.39
VIKING 1 ATMOS	45.4	35533.1	20.3	6.78
VIKING 2 ATMOS	32.7	31460.8	27.3	6.54
ΔPER = +2.78 km*	37.7	33844.7	12.3	6.65
ΔPER = -2.78 km*	37.8	33854.2	12.2	6.76
Δ ALPHA = +2.0°	37.9	33853.7	12.2	7.01
Δ ALPHA = -2.0°	37.4	33846.9	12.3	6.82
RMS OF DELTAS FROM NOMINAL	3.6	1585.7	9.1	0.26

* DELTA FLIGHT PATH ANGLE = ±0.10° (AT 125 KM)

** FINAL PARK ORBIT OF 250 X 33851 KM IS REACHED VIA ΔV1 AT APOAPSIS FOLLOWED BY ΔV2 AT PERIAPSIS (ΔV=ΔV1+ΔV2)

PROGRAMMATICS

Currently the AFE program is planned to better characterize aeroassist issues associated with non-equilibrium radiation, surface catalysis, and flowfield characterization via a flight experiment in the 1994 time frame. The flight data obtained will also act to validate CFD codes. This experiment will investigate the speed regime consistent with return from geosynchronous orbit (entry speeds of 9.6 km/s) of an Aeroassisted Space Transfer Vehicle (ASTV). The AFE flight test database will probably be sufficient to enable the development of an unmanned Lunar Evolution cargo vehicle. This vehicle could then be used to validate entry configurations for the subsequent manned vehicles. On the other hand, if the manned and unmanned flights are concurrent or if the advanced technology aerobrake concepts (such as flex/fabric) are used, a flight of will be required.

For manned planetary flights, one to two further flight tests will be required depending on the type of aerobrake configuration utilized. In all cases it would be highly optimistic to assume manned aerocapture at Mars could be undertaken without first accomplishing a precursor mission. Such a mission as the MRSR could fit this requirement though it would need AFE levels of instrumentation for suitable data return. Because such a mission would need to be completed before the start of serious manned Mars designs it would have to be completed (including data reduction) at least five to six years prior to launch. The current OEXP case studies looked at initial flights in the 2004 time frame, which would present a severe schedule driver. If high speed entry at Earth is maintained as a mission requirement (primarily driven by sprint mission and powered Mars abort options) it will probably necessitate a dedicated Earth capture flight test due to the very high radiation and leeside heating regimes encountered. Although the MRSR return segment could theoretically be boosted in energy to accomplish this mission, it is more likely that it would be carried out as a dedicated Earth orbital experiment in order to avoid jeopardizing the return of the MRSR's samples. Finally, if the use of the flex/fabric aerobrake concept is selected it will require a flight verification of the design issues mentioned earlier, probably replacing the high speed Earth capture test since the two requirements are mutually exclusive. Here again, for the baselined mission start dates used in OEXP case studies, the high speed Earth capture test and/or flex/fabric aerobrake flight test would have to be completed by 1999. This date is by no means technically impossible but it does require very aggressive funding to be achieved starting in the very early 1990's.

APPENDIX E. DETAILED REQUIREMENTS
SRD REQUIREMENTS AND SPECIFICATIONS ANALYSIS

Definitions:

Groundrules.	Broad rules set out in the SRD, either for overall application or to the particular Case Study
Requirements.	Detailed technical specifications given in the SRD to specify or constrain the Case Study.
Deviations.	Exceptions that must be taken to the SRD as written.
Derived Requirements.	Additional requirements that can be deduced from one or more (e.g., via combination of) SRD Requirements.
Assumptions.	Technical specifications not given in the SRD. Assumptions are generated by the TIA in order to conduct the Case Study.
Partials/Sensitivities.	Parametric variations about the selected point design, specified by SRD *or* TIA.
Options.	Changes from the baseline set of SRD requirements which uniquely define a Case Study, but are clearly specified in or inferred from the SRD as additional choices to be studied.
Alternatives.	Several different approaches to the same general objectives may be considered in implementing the Case Study. Each Alternative is generated by the TIA by replacement of one or several of the SRD requirements.

LUNAR EVOLUTION

Extracted from signed SRD dated March 3, 1989, Section 2.2 Lunar Evolution Case Study (pp. 7-47)

Groundrules:

- Achieve a test bed and learning center for long duration planetary missions (2.2.1.A)
- Develop a significant science research capability (planetary, astronomy, life sciences) (2.2.1.B, 2.2.2.G, H, 2.2.3.1.B)
- Develop resource potential of the moon (2.2.1.C, 2.2.2.F). Propellant production (2.2.3.1.H)
- Develop a lunar gateway for Moon and solar system (2.2.1.D). Mars evolution (2.2.2.K)
- Control ETO delivery per year to a specified level and determine capability at Moon (2.2.2.B)
- Reusable vehicles (2.2.2.I, 2.2.3.1.D)
  Exception: Maintain an expendable, post-TEI contingency direct Earth entry capability (2.2.4.2.2)
  Trans-lunar vehicles are assembled and serviced at Station Freedom (2.2.3.1.E)

Lunar ascent/descent vehicles are serviced and maintained on lunar surface (2.2.4.2.1.3)

- Annual IMLEO limit of 570 t/yr to 500 km gross, $\leq$ 90 t/yr dry (averaged over 2 yrs) (2.2.4.1.1)
- 3 phases: Outpost/Human-tended (crew 4, 6-mo. TOD)
  Experimental (crew 8, 2-yr TOD)
  Operational (crew of 8 up to 30, 2-yr TOD) (2.2.4.1.3)
- Commonality: minimum number of vehicles to fulfill functions of operational phase (2.2.4.2.1.2)
- No orbital nodes other than Space Station Freedom (2.2.4.4)
  Minimize requirements for on-orbit assembly, but
  make appropriate use of Freedom Station in time period 2004-8 (2.2.4.2.3.4.3).
  Freedom will *not* be used to store main-stage propellants [e.g., H/O] (2.2.4.3.2.8)
- Technology Level 6 for EAb by 1996, NEP by '06, LLOX production by '98, LH2, metals (2.2.4.1.4.1.L) by '08,
  1 MWe surface power by '98, lunar construction and transportation by '01 (2.2.4.9)

Requirements:

- ETO Cargo (2.2.4.8): Launches ≥45 day intervals, 4 launches/yr.
  140 t to i=28.5°, 500 km LEO. Accommodates 12.5 m diameter x 25 m long cargo load.
  Supports first mission in 2004.
  ETO Crew (2.4.4.8): 2 launches/yr. Not more than 3 days after a cargo launch.
  4 crew, 5 t cargo for lunar mission. 6 person servicing/repair crew plus 5 t cargo.
  Provide ferry in vicinity of Freedom [is this an OMV?] (2.2.4.8.1.2.3)
  Provide rescue capability (lunar STV) for 8 crew (2.2.4.8.2.1.4, -2.2.4)
- Program initialized in 2004 (2.2.2.D, 2.2.3.1.A, 2.2.4.1.2)
- Crew 4, growing to 30 (2.2.3.1.C) at base
  Minimum crew 2; Two are IVA/EVA proficient, two are EMT (2.2.4.1.6.1)
- Transportation vehicles shall be sized for pre-set ΔV's and durations (2.4.2.2.2.1, -3.1, and -4.1)
  Crew vehicles accommodate 8, mixed gender (2.2.4.2.2.2, -4.2), with 2 t cargo (-.3, -.3)
  LEO<—> LLO, Cargo of 20 t for initialization, TBD t for earth to LLO, 2014 (2.2.4.2.3.3)
  LLO<—> LSurf, Cargo of 20 t for all phases (2.2.4.2.5.3)
  LLOX, LSurf—>LLO. Amount of LLOX is 0.5 that needed for roundtrip
  of cargo or crew, LLO—>LEO—>LLO, initialization . TBD t for 2014 (2.2.4.2.6.3)
- Chemical propulsion, at initialization (2.2.3.1.G). H/O for crew (2.2.4.2.2.4.1, -4.4.1)
  H/O for cargo at initialization; <u>NEP</u> at technology enhancement (2014) (2.2.4.2.2.3, -3.4)
  H/O for LLOX transportation (2.2.4.2.6.4)
- Powered flyby aborts at moon (2.2.4.1.5)
- 30.3 d roundtrip flight time to moon (initial capability) (2.2.4.2.2.1.A, -3.1.A)
- Aerobraking at Earth for return vehicles (2.2.3.1.F, 2.2.4.2.2.4.2, -3.4.2).
  Entry speeds $\leq$ 11.5 km/s (2.2.4.1.5)
- Crew work/rest scheduling
  6 duty days/wk @ 8 hrs duty time/work day; 2 hrs/day exercise (2.2.4.1.6.2)
- Dual failure tolerant subsystems, operable from redundant locations (2.2.4.1.6.4.3)
  [<u>Note</u>: TIA assumes one engine out for cryopropellant-based propulsion and excludes this requirement for storable bipropellant-based systems]
- Protect against excessive cosmic and solar radiation.

Capability accessible within 30 minutes (2.2.4.1.6.4.2)
[Note: Only protection against solar flare radiation will be provided by TIA]
Radiation protection on MPV: provide 5 g/cm2 shielding (2.2.4.2.2.4.4)
[Assumption: does not include slant-path or astronaut mutual shielding benefits]

- All hazardous materials stored outside of pressurized elements (2.4.4.1.6.4.4)
- Isolation and rapid egress from any habitable element in emergency (2.4.4.1.6.4.5)
- All habitable elements shall have redundant escape paths (2.4.4.1.6.4.6)
  [but not necessarily rapid. See use of EVA escape path, under Derived requirements]
- Pressure integrity checks; adequate day/night lighting; pressure hatches (2.2.4.1.6.4.7-9)
- EVA-suited operation of emergency controls (2.2.4.1.6.4.10)
- 60 d safe-haven capability *on the lunar surface* (2.2.4.1.6.4.11)
- Para. 2.2.4.1.6.4.12 (p. 20). "sustain" mean "survive"? What does this requirement imply?
- Transport one injured crewmember back to Earth in addition to normal crew complement (2.2.4.1.6.4.13) [Note: This implies an 8-crew module actually has a 9-crew capacity]
- "Autonomous", on-board crew training capability (2.2.4.1.6.5.1)
- Technical procedures & Ops data electronically available at point of execution (2.2.4.1.6.5.2)
- Single crewmember maintenance, normal and contingency operation (2.4.4.1.4.6.1)
- Modular systems; spares (2.2.4.1.6.6.2-3)
- IVA/EVA systems comply with NASA Std 3000 (MSIS) (2.2.4.1.6.6.6)
- Prox Ops require direct operator viewing;
  areas requiring EVA access are viewable direct or by TV (2.2.4.1.6.6.4-5)
- Piloted rovers are pressurized if range ≥ 10 km (2.2.4.1.6.6.7); Crew ≥ 2 if range ≥ 1 km (2.2.4.1.6.8.4)
- EVA outside time <8 hrs; two crew minimum per EVA; all crew have personal suits (2.2.4.1.6.8)
- Propellant autonomous transfer in LEO, but astronaut backup of <40 hrs EVA, <100 hrs IVA (2.4.4.8.3.1)
- Rescue using ETO to LSurf (2.2.4.8.2.1.4, -2.2.5)
- Communications (2.2.4.1.7, including Table 2.2.4.1.7.2-1)
- Surface
  Base is on equator at Mare Tranquillitatis (24° E); farside observatory at 141° E on equator (2.2.4.5.1)
  Mass allocations are 105 t, 260 t, and 280 t for Outpost, Experimental, and Operational phases (2.2.4.5.3)
  Base will service, maintain, and store all launch vehicles (2.2.4.5.6)

Deviations from SRD:

Dual failure tolerant subsystems, operable from redundant locations (2.2.4.1.6.4.3)

TIA will assume one engine out for cryopropellant-based propulsion and excludes this requirement for storable bipropellant-based systems.

Lunar LOX Utilization

Greatest payoff is determined to utilize LLOX just for LSurf <—> LLO and return-to-Earth from LLO. Amount of LH2 to be carried is just sufficient to provide these capabilities.
No net return of LLOX to LEO.

Derived Requirements:

- 1.2 t radiation shielding per 4 crew (for LPV only) (2.2 t for 8 crew cab)
- Venting of modules and EVA as one of the two escape paths is permissible (private communication with D. Bland, 2-22-89; see also 2.4.4.1.6.4.6)
- Optimize O/H mixture ratio against IMLEO for use of LLOX
- LCSV is only partially-loaded when used for crew 4
- All landers are Lunar surface-based [early landers expendable?]

MARS EVOLUTION

Extracted from signed SRD dated March 3, 1989, Section 2.3 Mars Evolution Case Study (pp. 49-98)

Groundrules:

- Establish a Martian moon "gateway", followed by a Mars surface facility (2.3.1.A, C)
- Significant science research capability (2.3.1.B)
  Explore *both* Martian moons (2.3.4.1.6.1.A)
- Reusable transportation system (2.3.4.2.2)
  But aerobrake for A/C only at Mars for cargo (2.3.4.2.3.4.2) (implies expendable)
  Expendable ECCV for contingency direct entry (2.3.4.2.2)
- Commonality: Minimum number of vehicles (2.3.4.2.1.2)
- All-up, split, or convoy missions allowed (2.3.4.1.7.A)
  Landings shall be in daylight (2.3.4.1.7.D, 2.3.4.2.4.4.2.A.i, 5.4.2.A.i)
- Space Station Freedom support (2.3.4.3)
  5, 6, and 7 crew (phased); 10 t cargo; 15 month vehicle processing time
- LEO Node (2.3.4.4)
  Provides a free-flyer spacecraft for man-tended LEO assembly/checkout, including:
  Mating/assembly, construction, deployment/retrieval, on-orbit checkout, debris protection.
- Technology Development (2.3.4.9): Technology Level 6 for MAb and EAb by 1996, NEP by 2005, NTR by 2007

Requirements:

- Human mission departs Earth in 2004 (2.3.4.1.2)
  Crew ≥ 3. Two are IVA/EVA proficient, two are EMT (2.3.4.1.8.1)

- Development Phases for Martian surface base (2.3.4.1.2)
  Science outpost: instrumentation
  Human-tended: 5 crew, 1-yr TOD. Not permanently occupied.
  Operational: 7 crew, 2-yr TOD. Global access to Mars. Use indigenous resources for life support.
- H/O propellant production at the gateway (2.3.4.1.4)
- Tethers: Facility for momentum exchange and propellant transfer at the gateway.
  Determine applications and assess advantages. (2.3.2.F, 2.3.4.1.4, 2.3.4.5.2.2, 2.3.5.2.E)
- Personnel Transportation
  LEO <—> Gateway

Accommodate 5 crew, mixed gender. Increase to 7 crew in 2014 (2.3.4.2.2.2)
Accommodate a Mars descent/ascent vehicle (5-crew capacity) plus 20 t add'l cargo (2.3.4.2.2.3)
Gateway <—> MSurf
Land to elevations up to 15 km (2.3.4.2.4).
5 crew, mixed gender, increased to 7 crew in 2014 (2.3.4.2.4.2)
Include 10 t cargo, Gateway —> MSurf (increase to 25 t if no ascent cab included) (2.3.4.2.4.3)

- Cargo Transportation (2.3.4.2.3.3)
  LEO —> Gateway: 150 t equipment , enhanced to TBD in 2014.
  Gateway —> MSurf: 50 t equipment (of the 150 t above), to TBD in 2014. (2.3.4.2.5.3, -2.3.3)
- 1252 d flight time design (for Mars flyby abort); up to 400 d at Mars (2.3.4.2.2.1)
- Artificial-g spaceship, ≥1/3 gee, ≤4 rpm (2.3.4.2.2.4.3, but not on ascent/descent vehicles, 2.3.4.2.4.4.3)
  Determine operational limits of coriolis forces (2.3.4.6.2.4)
- Aerocapture at Mars and Earth for personnel vehicle (2.3.4.2.2.4.2), at Mars for cargo (2.3.4.2.3.4.2)
  No restriction on aerobrake L/D ratio.
  Mars entry velocity ≤9.5 km/s (2.3.4.1.7.E). (No limit on max-deceleration)
  Earth entry velocity ≤13.5 km/s (2.3.4.1.7.G). (No limit on max-deceleration)
  Mars Descent Vehicle includes aerobrake to lower apoapsis to circularize for lighting control (2.3.4.2.4.4.2, -5.4.2)
- Transportation vehicles shall be sized for pre-set ΔV's and durations (2.3.4.2.2.1, -3.1, and -4.1)
  Requirements change after 2014 for personnel carrier (2.3.4.2.2.1.B; to accommodate NTR, 2.3.4.2.2.4.1.B)
- Chemical propulsion initially for both cargo and human transportation (2.3.4.2.2.4.1, -3.4.1)
  Nuclear Thermal Rocket (NTR) for MPV in 2014 (2.3.4.2.2.4.1.B)
  Nuclear Electric Propulsion (NEP) for MCV in 2014 (2.3.4.2.3.4.1.B)
  All ascent/descent vehicles remain chemical propulsion at all times (2.3.4.2.4.4.1, -5.4.1)
- Multi-impulse TMI and TEI are permitted (2.3.4.1.7.B, -.F) (to minimize gravity and plane change losses)
- No single-point failure in subsystems of safety-critical systems (2.3.4.1.8.4.3)
  Dual failure tolerant subsystems, operable from redundant locations (2.3.4.1.8.4.3)
  [Note: TIA assumes one engine out for cryopropellant-based propulsion and excludes this requirement for storable bipropellant-based systems]
- Emergency operation of all systems by EVA-suited crewmember (2.3.4.1.8.4.10)
- Protect against excessive cosmic and solar radiation.
  Capability accessible within 30 minutes (2.3.4.1.8.4.2)
  [Note: No protection against cosmic radiation will be provided by TIA]
  Radiation protection on MPV: provide 5 g/cm2 shielding (2.3.4.2.2.4.4)
  [Assumption: does not include slant-path or astronaut mutual shielding benefits]
  [Assumption: provided only in a radiation storm shelter, not for entire hab module]
- Crew work/rest scheduling
  6 work days/wk @ 8 hrs duty time/work day, 2 hrs/day exercise (2.3.4.1.3.2.2,-.3)
  Until permanent human presence is established, all crew have same day off (2.3.4.1.8.2.4)
- All hazardous materials stored outside of pressurized elements (2.3.4.1.8.4.4)

- Isolation and rapid egress from any habitable element in emergency (2.3.4.1.8.4.5)
  All habitable elements shall have redundant escape paths (2.3.4.1.8.4.6)
  [Note: but not necessarily rapid. See use of EVA escape path, under Derived requirements]
- Pressure integrity checks; adequate day/night lighting; pressure hatches (2.3.4.1.8.4.7-9)
- "Autonomous", on-board crew training capability (2.3.4.1.8.5.1)
- Single crewmember maintenance, normal and contingency operation (2.3.4.1.8.6.1)
  Modular systems; spares (2.3.4.1.8.6.2-3)
  Technical procedures & Ops data electronically available at point of execution (2.3.4.1.8.5.2)
- IVA/EVA systems comply with NASA Std 3000 (MSIS) (2.3.4.1.8.6.6)
  Propellant transfer ops require ≤100 p-hr IVA, ≤ p-hr EVA (2.3.4.8.3.1)
- Prox Ops require direct operator viewing;
  areas requiring EVA access are viewable direct or by TV (2.3.4.1.8.6.4-5)
- EVA outside time <8 hrs; two crew minimum per EVA; all crew have personal suits (2.3.4.1.8.8)
  All rover excursions ≥ 1 km require 2 crew (2.3.4.1.8.8.4). ≥ 10 km require pressurized rover (2.3.4.1.8.6.7)
- Communications (2.3.4.1.9.2, including Table 2.3.4.1.9.2-1)
  10 Mbps MTE, 20 Mbps ETM at 2.5 AU
  [Note: 10 Mbps MTE exceeds estimated need]
- User requirements (Instrument Packages): Solar; Cosmic Dust; Cosmic Rays; Gamma Bursts; Biomedical. Engineering characteristics provided in Study Data Book (2.3.4.2.2.5).
- ETO Cargo (2.3.4.1.1): Launches ≥45 day intervals, 4 launches/yr.
  140 t to i=28.5°, 500 km LEO. Accommodates 12.5 m diameter x 25 m long cargo load.
  570 t/yr to 500 km. Dry to LEO ≤180 t per two *consecutive* years (2.3.4.1.1)
  ETO Crew (2.3.4.8.2.2.3): Mission crew of 5 to Freedom, plus 5 t cargo; servicing crews to 6.
- Planet Surface System Requirements: Section 2.3.4.5.
  Landing site is Chryse basin complex (equator, 33.5° W).
  Phobos/Deimos Surface Base mass allocation: 100 t (2.3.4.5.2) (for PhLOX, PhLH2 production)
  Mars Base mass allocations: Outpost, 35 t; Human-tended, 120 t; Operational, 150 t (2.3.4.5.1.3)
  ISRU for Mars H2O, MLOX, MLH2, construction. (2.3.4.5.1.5)
  Provide surface transportation, including shielding against GCR and SPE (2.3.4.5.1.7)
  Mars Target Science:
  Geology, Geophysics, Atmospheric, Particles and Fields, Exobiology, Resource Assessment
  Mars Platform Science (Laboratory): Geochem/Petrology/Paleomag, Biochemistry, Life Sciences
  Mars Surface Science: Sample return, mobile vehicle, sampling, automated geophys/atmo stations

Deviations:

- Dual failure tolerant subsystems, operable from redundant locations (2.3.4.1.8.4.3)
  TIA will assume one engine out for cryopropellant-based propulsion and excludes this requirement for storable bipropellant-based systems
- MCV is Expendable
- MCSV is Reusable (post Gateway Operational phase)
- TMIS stages expended (make up part of MPV and MCV)
- ΔV of 200 m/s post-EOC not included.
  STV assumed to accomplish this

Derived Requirements:

- Provide windows (to view space and Mars surface non-electronically, 2.3.4.1.8.3.2)
- Venting of modules and EVA as one of two escape paths is permissible (private communication with D. Bland, 2-22-89; see also 2.3.4.1.8.4.6)
- Vehicles must operate autonomously during non-critical periods [48 work-hrs/week, leaving 120 hrs/week (71% of time) when no crew are on duty (2.3.4.1.8.2)]
- Dual habitation modules required on MPV and on MSurf

MARS EXPEDITION

Extracted from signed SRD dated March 3, 1989, Section 2.4 Mars Expedition Case Study (pp. 99-127)

Groundrules:

- Achieve a human landing on Mars ASAP ("earliest feasible mission opportunity", 2.4.2.B, 2.4.3.1.A)
- One human mission only (2.4.4.1.3)
- All space transfer vehicles are expendable (2.4.4.2.2.,-3)

- Transportation system to be launched *intact*, with no assembly in LEO (2.4.4.2.2.4.5, -3.4.3; 2.4.3.1.B) But both support propellant transfer in LEO (2.4.4.2.2.4.5, -3.4.3)
- No orbital nodes are required (2.4.4). Space Station Freedom provides LSS qualification, but no unique capabilities or accommodations (2.4.4.3)
- 1995 technology (2.4.4.1.3) (but "allowing for very high leverage technology extensions", 2.4.4.1.3) Technology Level 6 for MAb by 1994, ECCV brake by '96, Mars landing Nav and Hazard Avoidance by '95 (2.4.4.9)
- Maximum use of orbiters and landing beacons from precursor mission(s) (2.4.4.1.3)
- Aggressive Phase C/D schedules (4-5 yrs) (2.4.4.1.2)

Requirements:

- Split/sprint trajectory (2.4.4.1.1.A), with free return abort for piloted vehicles (-.C)
- Human transportation vehicle is operational by July 2002 (2.4.4.2.2)
  Vehicle sized for a single mission only (2.4.4.1.3)
  Crew of 3 (2.4.4.2.2.2, 2.4.3.1.E); Two are IVA/EVA proficient, two are EMT (2.4.4.1.4.1)
  No cargo capacity on MPV (2.4.4.2.2.3)
- Cargo transportation vehicle is operational by March 2001 (2.4.4.2.3)
  Cargo capacity is the MDV, plus 10 t additional equipment with TBD dimensions (2.4.4.2.3.3, -4.3)
- Zero-g spaceship (2.4.4.2.2.4.3)
- Mars aerocapture (2.4.3.1.H). [Note: Differs in this respect from CS-1.0 of FY88]
  Aerobrakes of L/D between 0.9 and 1.2 for both cargo and piloted vehicles (2.4.4.2.2.4.2, -3.4.2)
  Mars entry velocity ≤9.5 km/s (2.4.4.1.1.E). Max-deceleration ≤5 gee (-.H)
- Direct entry at Earth (2.4.3.1.H).
  Note: Not a *requirement*. This appears only in 2.4.3, Ref. Mission. TIA accepts Direct Entry as baseline, however.
  Earth entry velocity ≤16.0 km/s (2.4.4.1.1.G). Max-deceleration ≤5 gee (-.H)
- Transportation vehicles shall be sized for pre-set ΔV's and durations (2.4.4.2.2.1, -3.1, and -4.1)

- 730.3 d flight time design; 30 d at Mars (2.4.4.2.2.1) [Note: 730 d is much greater than "sprint" times]
- Chemical propulsion for both cargo and human transportation (2.4.4.2.2.4.1, -3.4.1)
- Multi-impulse TMI and TEI are permitted (2.4.4.1.1.C, -.F) (to minimize gravity and plane change losses)
- MDV uses storable propellants; aerobraking; no rad protection (2.4.4.2.4.4.1-4)
  MDV can land at altitudes to +5 km (2.4.4.2.4). Landing must occur in daylight (2.4.4.1.1.D)
  All 3 crewmembers to Mars surface for 20 days (2.4.4.2.3.3, -4.2)
  10 t cargo with TBD dimensions (2.4.4.2.4.3)
    Note: -2.4 reads "transfer crew and cargo from ... orbit to ... surface and back to ... orbit"
    TIA assumption is that transportation of 10 t cargo from surface back to orbit is *not* required
- No single-point failure in subsystems of safety-critical systems (2.4.4.1.4.4.3)
  Dual failure tolerant subsystems, operable from redundant locations (2.4.4.1.4.4.3)
    [Note: TIA assumes one engine out for cryopropellant-based propulsion and excludes this requirement for storable bipropellant-based systems]
- Emergency operation of all systems by EVA-suited crewmember (2.4.4.1.4.4.10)
- Protect against excessive cosmic and solar radiation.
  Capability accessible within 30 minutes (2.4.4.1.4.4.2)
    [Note: No protection against cosmic radiation will be provided by TIA]
  Radiation protection on MPV: provide 5 g/cm2 shielding (2.4.4.2.2.4.4)
    [Assumption: does not include slant-path or astronaut mutual shielding benefits]
    [Assumption: provided only in a radiation storm shelter, not for entire hab module]
- Crew work/rest scheduling
  6 duty days/wk @ 8 hrs duty time/work day, 2 hrs/day exercise (2.4.4.1.4.2)
- All hazardous materials stored outside of pressurized elements (2.4.4.1.4.4.4)
- Isolation and rapid egress from any habitable element in emergency (2.4.4.1.4.4.5)
  All habitable elements shall have redundant escape paths (2.4.4.1.4.4.6)
    [Note: but not necessarily rapid. See use of EVA escape path, under Derived requirements]
- Pressure integrity checks; adequate day/night lighting; pressure hatches (2.4.4.1.4.4.7-9)
- "Autonomous", on-board crew training capability (2.4.4.1.4.5.1)
- Single crewmember maintenance, normal and contingency operation (2.4.4.1.4.6.1)
  Modular systems; spares (2.4.4.1.4.6.2-3)
  Technical procedures & Ops data electronically available at point of execution (2.4.4.1.4.5.2)
- IVA/EVA systems comply with NASA Std 3000 (MSIS) (2.4.4.1.4.6.6)
  Propellant transfer ops require ≤100 p-hr IVA, ≤ p-hr EVA (2.4.4.8.3.1)
- Prox Ops require direct operator viewing;
  areas requiring EVA access are viewable direct or by TV (2.4.4.1.4.6.4-5)
- EVA outside time <6 hrs; two crew minimum per EVA; all crew have personal suits (2.4.4.1.4.8)
- Communications (2.4.4.1.5, including Table 2.4.4.1.5.2-1)
  10 Mbps MTE, 20 Mbps ETM at 2.5 AU. "Continuous communication is *not* required"
  [Note: 10 Mbps MTE exceeds estimated need. Range of MPV-to-Earth never exceeds 1.8 AU]
- User requirements (Instrument Packages): Solar; Cosmic Dust; Cosmic Rays; Biomedical.
  Engineering characteristics provided in Study Data Book (2.4.4.2.2.5). Minimal science equipment (2.4.3.1.F)

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